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U. S. ARMY

TRANSPORTATION RESEARCH COMMAND

FORT EUSTIS, VIRGINIA

EXPERIMENTAL RESEARCH

THEORY, DEVELOPMENT, AND TEST OF A CRASH FIRE-INERTING SYSTEM FOR RECIPROCATING ENGINE HELICOPTERS

December 1963

Contract DA 44-177-AMC-888(T) *TRECOM Technical Report 63-49*

prepared by :

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Postcrash fire contributes significantly to the fatality toll in Army aircraft accidents. A prime source of ignition for these fires is the aircraft engine and its exhaust system. The fire inerting system discussed in this report was designed, fabricated and tested by AVSER in an effort to alleviate the occurrence of fire in aircraft equipped with reciprocating engines.

Future crash fire prevention programs will consider problems peculiar to turbine engine aircraft and will investigate other methods of combating the crash and postcrash fire hazard.

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AvSER 63-1

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SUMMARY

This report describes the theory, development, and test of a fire-inerting system for helicopters powered with reciprocating engines. The feasibility of constructing a lightweight system capable of operating under typical crash conditions is demonstrated. Areas requiring additional development are discussed.

CONCLUSIONS

It is concluded that:

1. A fire-inerting system for helicopters should be in operation within 0.20 second or less after first impact.

2. A "time to operate" of 0.20 second can be attained in an operational fire-inerting system.

3. A fire-inerting system can be installed in a 10,000-pound aircraft with as little as 80 pounds increase in overall weight including coolant.

4. The construction and installation into service aircraft of a combination manual-automatic fire-inerting system meeting the above weight and "time-to-operate" limitations is possible with the present state of the art.

RECOMMENDATIONS

It is recommended that:

1. Continuance of this study be made with the turbine engine, since the primary source of power in many present day and most future production helicopters is the turbine engine.

2. An operational electrical de-energizing system be developed. This system must provide for safety while in normal flight, yet should be simple and afford maximum protection to the aircraft occupants in a crash.

3. A simple but reliable operational fire-inerting-system triggering device or initiator be developed.

4. Methods of eliminating abraded sparks which have sufficient intensity to provide ignition be sought and immediately employed in all new aircraft designs.

5. The feasibility of using a less-flammable hydraulic fluid be investigated.

6. Techniques that will improve fuel containment be investigated both theoretically and experimentally.

7. The feasibility of utilizing special surfacing to increase the surface ignition temperature in operational systems be evaluated. For example, when metals are surfaced with certain compounds, such as platinum, their temperature must be raised an additional 300 to 700 degrees F. to provide ignition (reference 14).

8. A study of the nature of fuel spillage patterns in helicopter accidents, and their effect upon ignition source and ignition delay, be made to provide a better understanding of the helicopter postcrash fire problem.

INTRODUCTION

Postcrash fire continues to be one of the most significant factors affecting survival in aviation accidents experienced by the U. S. Army, particularly with rotary-wing aircraft. Information provided in reports prepared by the United States Army Board for Aviation Accident Research (USABAAR) (references 1 and 2) indicates that the problem of survival in accidents involving postcrash fire is very significant in fixed-wing accidents and is probably the most critical factor in rotary-wing accidents. Information provided in these reports concerning accidents which occurred during the period 1 July 1957 through 30 June 1960 indicates that approximately 63 percent of all fatalities occurred in rotary-wing accidents which experienced fire. Thirty-five percent of the fatalities experienced in fixed-wing aircraft occurred in accidents involving fire.

Because of the damage resulting from impact loads, it is frequently difficult to determine the exact cause of fire; however, on the basis of the information available, it is indicated that 75 to 80 percent of the fires experienced resulted from ruptured fuel tanks or fuel lines. It is apparent that a reduction in the occurrence of fuel spillage near potential ignition sources could be instrumental in preventing many of the postcrash fires and the resultant fatalities.

Several approaches are presently being taken to resolve this problem. One employs a breakaway fuel tank concept, another utilizes crash-resistant flammable fluid systems, and another involves the solidifying of the fuel within the tank during impact. This report concerns still another approach: ignition source inerting.

Aviation Safety Engineering and Research (AvSER) was awarded a contract by U. S. Army TRECOM to study the feasibility of developing a crash-fire-inerting system for Army helicopters. Although several agencies had worked on the fixed-wing crash-fire problem, no organizations had conducted research aimed at reducing the incidence of crash-fires in helicopter accidents. After a thorough review of many fire research reports, researchers at AvSER were well aware of the known ignition sources and fuel spillage problem areas which had previously been reported (see References). Using this information and other data obtained through laboratory and field tests, AvSER designed a fire-inerting system, which was dynamically tested in a full-scale crash of an H-21 helicopter on 12 September 1962. The research and development done by AvSER in connection with this fire-inerting system is discussed in the following sections of this report.

ANALYSIS OF THE PROBLEM

Fires occur only when three basic elements are present: a fuel, an oxidizer, and an ignition source. Postcrash control of all three elements would be ideal; however, complete control of the oxidizer (the atmosphere in the crash area) does not appear feasible. Local control of the oxidizer at specific ignition points is possible as shown in the work reported by NACA and also by Walter Kidde and AvSER. Fuel control in the form of fuel containment can effectively reduce fire hazard and fire volume when leaking fuel is ignited by uncontrolled sources such as static or friction sparks. The ignition source appears to be the most feasible of the three elements to control.

The literature search which preceded the experimental testing included a thorough evaluation of the work done on postcrash fire prevention in fixed-wing aircraft at the NACA Lewis Laboratories during 1949-1957 and the subsequent work done by the Walter Kidde Company for the Air Force in 1953-1958. The NACA tests clearly demonstrated that ignition source inerting systems capable of preventing fires in approximately 80 to 90 percent of the potentially survivable fixed-wing accidents could be based upon the control of only two ignition sources: (1) the aircraft electrical system, and (2) the aircraft exhaust system.

Past research by other agencies, including the NACA's fixed-wing crash-fire program (references 8, 9, and 14), has shown the ignition sources to be:

1. Hot surfaces.

2. Friction and chemical sparks.

3. Engine exhaust flames.

4. Engine induction flames.

5. Electric arcs.

6. Static sparks.

Other considerations made apparent from the study of previous reports are:

1. Of the four flammable fluids carried in aircraft (gasoline, hydraulic oil, lubricating oil, and alcohol), from the standpoint of survival, gasoline is the most dangerous, even though it ignites at a considerably higher temperature than the oils.

2. A fire cannot be started from a hot-surface ignition source if the hot surface is surrounded by an inert atmosphere.

3. Removing the battery, the inverters, and the generator system from the circuit is an effective way to eliminate the electrical-spark ignition source.

4. Engine flames can best be eliminated by leaving the engine ignition on and allowing the engine to burn the ingested fuel normally. By cutting the fuel and rapidly "choking off" the engine with CO_2, the internal portion of the engine can be inerted as engine- and aircraft-breakup takes place.

5. Changes in the surface composition of the exhaust duct material may effectively increase the temperature to which the surface must be heated to induce ignition. Certain surfaces such as platinum must be heated to temperatures of 300 to 700 degrees F. greater than for stainless steel to ignite flammable mixtures (reference 14).

In the application of fire-inerting systems to any aircraft, the time factor is of utmost importance; that is, the system must be activated and must be providing satisfactory inerting at the earliest time at which ignition could normally take place. As a result of the NACA tests, it can be concluded that ignitions initially occur in fixed-wing aircraft approximately 1 second or more after impact. On the basis of impact tests conducted with three H-25 and two H-21 helicopters by AvSER, there is considerable evidence that the time to ignition for accidents involving rates of descent in the order of 30 to 40 feet per second may be as little as 0. 20 second. (See Timing of Fire Inerting System under the heading, Experimental Procedure - Preliminary Testing, page 22.) This low time to ignition is associated with the close proximity of the fuel cells to the ignition sources in the helicopter. It should be observed that impact at 30 to 40 feet per second vertical descent would be considered potentially survivable from the crash injury point of view; but in helicopter accidents, massive fuel spillage and vaporization close to ignition sources greatly increase the danger of fire.

Obviously, a fire-inerting system is practical only if it provides inerting within the time limits imposed by the flight characteristics and configuration of the specific aircraft in which it is installed. In view of the apparent 0.20-second time constant for the H-25 (and H-21) helicopters, AvSER has concentrated its present efforts upon the development and testing of a system applicable to an H-21 aircraft. Emphasis was placed on the inerting of the engine and engine exhaust system, under the hypothesis that the electrical system can readily be inerted in 0.20 second. This was demonstrated as a side experiment in the subsequent test of the engine system, although no effort was made to develop a prototype electrical inerting system suitable for installation on a service aircraft. This is considered to be a routine developmental project.

DESIGN OBJECTIVES

The basic requirements established for the experimental engine inerting system were:

1. To cool all hot surfaces to a temperature below the ignition temperature of flammable fuels and oils.

2. To prevent combustible mixtures from contacting hot surfaces until all such surfaces have been cooled below the ignition temperature of the mixtures.

3. To inert the interior of the engine, from the induction system to the exhaust system outlets so that flames do not appear at the engine air intake or exhaust.

4. To shut off fuel and oil supply at the engine.

5. To provide the above functions with a weight limitation of 100 pounds and a time-to-inert of 0. 20 second or less when subjected to crash environments typical of a severe, but potentially survivable accident.

The approaches used by AvSER to accomplish each of these objectives will be discussed individually. Many experiments, some of which are discussed in this report, were conducted to establish the specific requirements of the system described above.

COOLING OF HOT SURFACES

By the use of a pyrometer, thermocouples, and pyrometric paints, the maximum operating temperature of various heated components of an operating helicopter engine was obtained. Next, the highest temperature to which the flammable fluids could be exposed without ignition was determined. Based upon the literature study and tests conducted by AvSER, 400 degrees F. was chosen as the safe temperature to which the hot surfaces must be cooled (see Appendix II). It was found that, for the H-21, the exhaust collector ring was the only item which required cooling. Water was selected as the exhaust collector ring coolant in this study. Obviously, for cold-weather operation, suitable additives will be required to lower the freezing point, or other coolants must be used.

The actual quantity of water needed was determined by the mass and initial temperature of the hot components. The coolant flow rate (which is directly related to the area of the hot surface) is governed by the rate at which the water can be converted to steam. The ideal situation, from the standpoint of weight and efficiency, would be to discharge the water spray at the exact rate that it could be converted to steam. However, as a safety margin the coolant must be discharged at a slightly higher rate to allow for spill-off and inefficient spray coverage.

In order to hasten the development of an inert atmosphere, gaseous nitrogen was mixed with the water and distributed through the spray tubes to the exhaust collector ring. Thus, with the steam and nitrogen being discharged simultaneously, an inert atmosphere was created more rapidly than if water alone had been used.

The introduction of nitrogen into the coolant also serves two additional functions:

1. It propels the water droplets to the spray-tube extremities and maintains a more constant pressure throughout the distribution system (reference 6).

2. It permits the use of larger spray discharge outlets, thus helping to eliminate the spray outlet clogging problem (reference 6).

The nitrogen-water mixture was carried to the desired locations through spray distribution tubes located between the exhaust collector ring and the shroud (Figures 1A, 1B).* The shroud is an effective retainer of the water spray and of the steam formed therefrom; however, between the engine cylinder-head exhaust outlet and the point where the individual exhaust pipes enter the shroud, there are nine unshrouded sections of the exhaust manifold, each approximately 4 inches long. It was determined by testing that the most effective way to retain water droplets in this area and to insure better wetting (thus, the best cooling) was to wrap the exposed exhaust pipe with 80-mesh stainless steel wire screen

* In the H-21 helicopter, the exhaust collector ring is shrouded. The clearances between the exhaust collector ring and the shroud vary from 1/4 inch to 3/4 inch. A cooling fan in front of the R1820-103 engine forces air between the exhaust collector ring and the shroud.

pressed into a waffle pattern (Figure 2). Because the waffle pattern allows the water to flow back and forth under the individual channels created by the waffle grid, the coolant spill-off is reduced. According to earlier studies, it was determined that the 80-mesh screen most effectively allowed the vaporized steam to escape and yet retained much of the water (reference 6).

Figure 1A. Spray Distribution Tube Layout
With the shroud removed, the spray tube layout and spray manifold are visible on the actual test aircraft (postcrash photo).

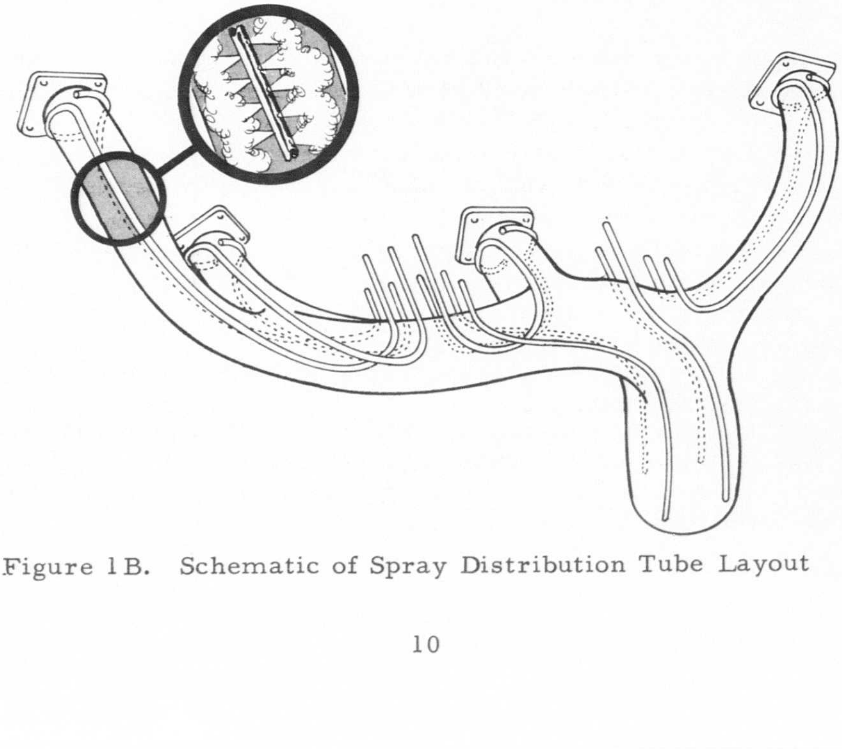

Figure 1B. Schematic of Spray Distribution Tube Layout

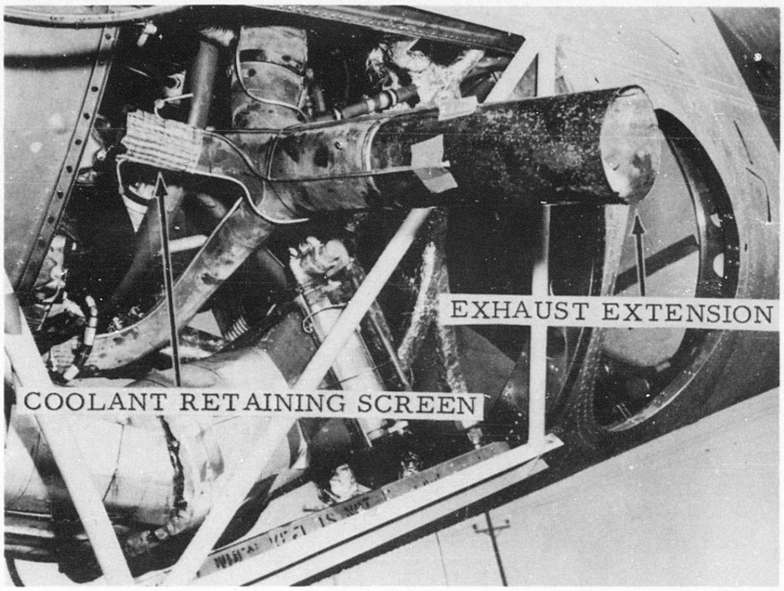

Figure 2. Coolant Retaining Screen.
Early spray tube layout is shown on the exhaust collector ring of the test bed aircraft. The shroud has been removed for the photograph.

The size of the coolant distribution tubes was based upon the smallest practical size that would insure satisfactory flow (reference 6). The spray distribution tubes were contoured to the exhaust collector ring in a manner that would insure a complete wetting of the collector ring surface with minimum restriction of the normal airflow through the shroud.

The diameter of the nitrogen-water spray orifices in the distribution tubes was 0.022 inch. This was the smallest diameter hole that experience had shown as being acceptable from the standpoint of clogging. The tubing eventually selected for the spray distribution system was .180-inch I. D. stainless steel. The number of spray orifices per individual distribution tube was determined on the basis of best coolant coverage without reducing nitrogen-water flow at the orifices toward the ends of the tubes (reference 6).

There were 24 spray distribution tubes on the R1820-103 radial engine exhaust system, 12 tubes per side. The distribution tubes were attached to a spray distribution manifold (12 tubes per manifold). The spray manifold was coupled directly into a nitrogen-water aspirator or mixing chamber as shown in Figure 3 A.

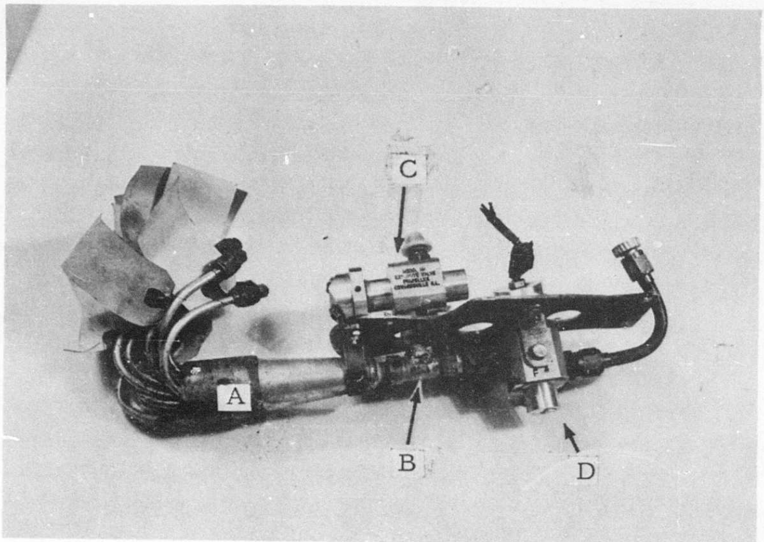

Figure 3A. Spray Distribution Manifold.
Details identified by arrows are (A) aspirator (B) mixing chamber (C) explosive-actuated valve for the water (D) explosive-actuated valve for nitrogen.

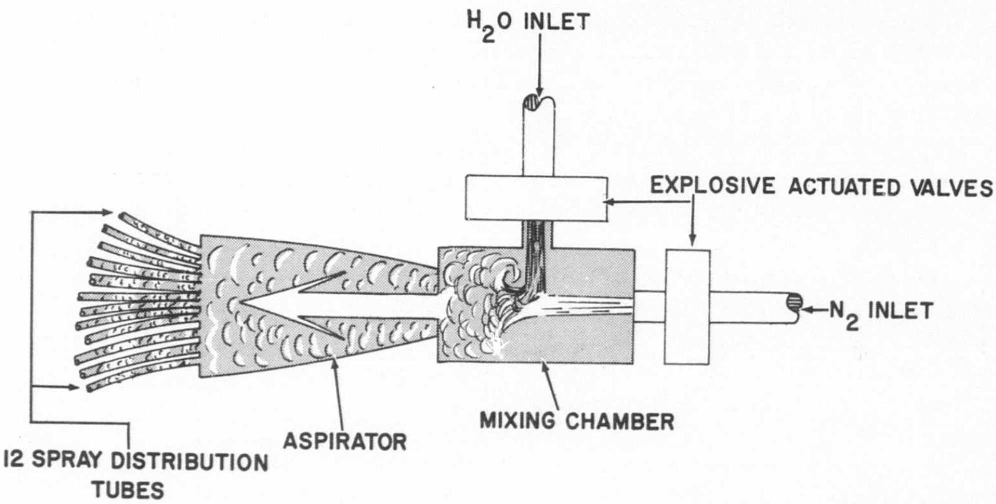

Figure 3B. Schematic of Spray Distribution Manifold.

As a result of extensive testing (see Experimental Procedure, page 24), it was determined that the diameter of the two orifices in the aspirator should be 0.055 inch for the nitrogen and 0.082 inch for the water. The best driving pressure was determined to be 200 psi maintained during actual operation.

The quantity of water needed to cool one-third of the exhaust manifold to 400 degrees F. was determined experimentally to be 3 quarts; thus, for the entire engine manifold, 9 quarts were required. Actually 10 quarts were used. The extra quart was provided for a margin of safety. The quantity of nitrogen needed was determined to be that contained in a 200-cubic-inch pressure vessel at 2,000 psi.

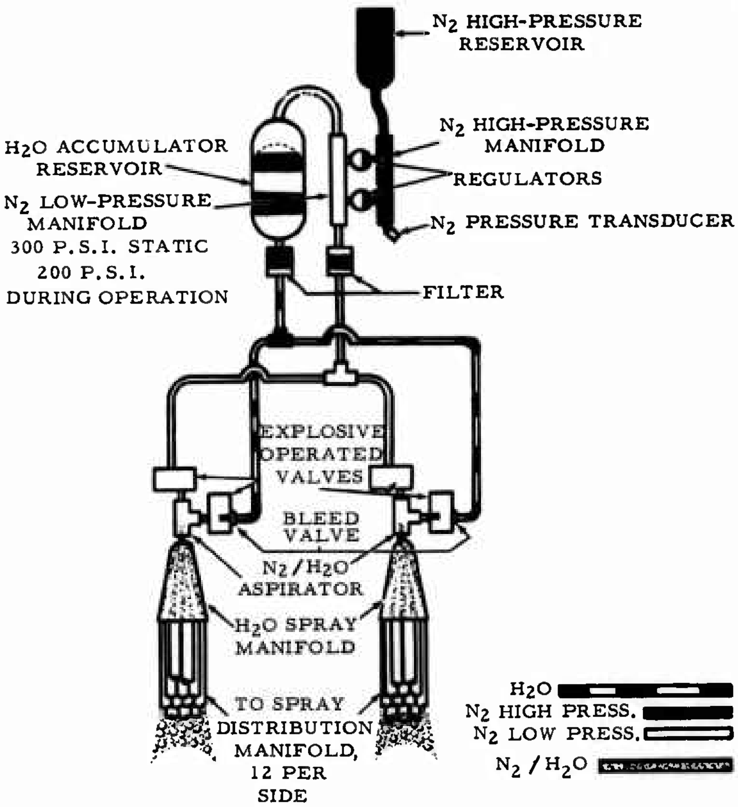

Figure 4. N_2-H_2O Spray System Schematic.

Figure 4 shows a schematic of the system. It can be observed that the

high pressure nitrogen passed through two parallel regulators and into a low pressure (300 psi static, 200 psi during operation) manifold. One end of the N_2 low-pressure manifold connected via flex tube to the aspirator. The other end of the manifold, which connected to the "air" side of a 2-1/2-gallon-capacity accumulator, provided the driving pressure for the water.

A positive flow of the water, regardless of the reservoir orientation, was assured because an accumulator has a flexible diaphram that separates the nitrogen from the water. From the accumulator, water was carried via flex line to the aspirator.

Filters were incorporated in the system to eliminate the possibility of foreign particles interfering with the operation. To eliminate any time delay that would result due to an air space in the water-line, air-bleed valves were used to allow complete filling of the lines with water prior to the test. Explosive actuated valves were used to start the water and nitrogen flow into the aspirator. Figure 5 shows a sketch of the valve.

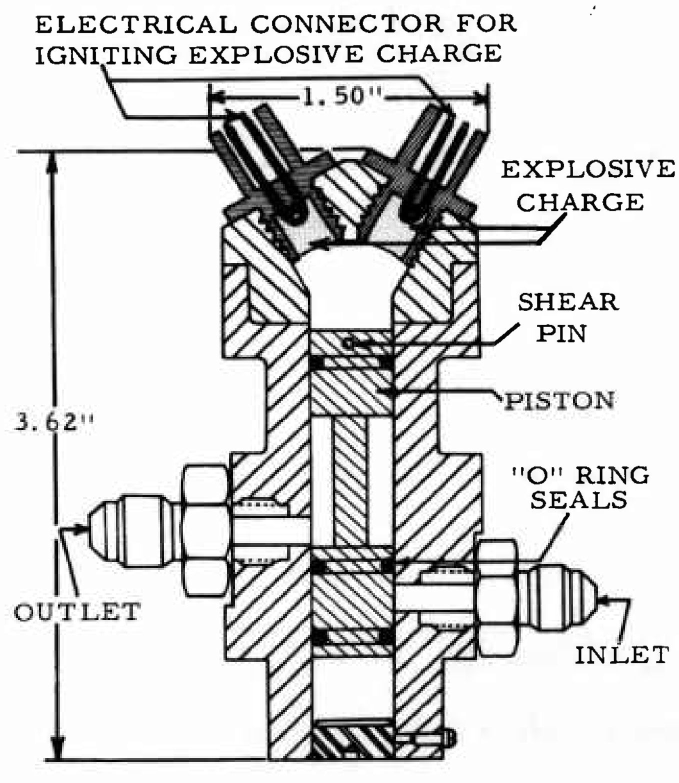

Figure 5. Explosive-Operated Valve.

Care was taken to insure that all the valves and aspirator-manifold components were placed well within the confines of the engine itself to protect them from crash damage. If these items were not protected by the engine and the engine should move through the aircraft during the crash, the various components could be damaged and the system rendered useless.

Figure 6 shows how the spray manifold was mounted on an intercylinder deflector located between two cylinders on the radial engine.

The water and nitrogen reservoirs and the two regulators were mounted on the inside of the engine compartment near the engine. A flex line connected the reservoirs to the explosive valves. This allowed for movement of the engine within the airframe without interrupting the coolant or nitrogen flow.

INTERNAL INERTING SYSTEM

While the steam and nitrogen from the water-spray system effectively inert the atmosphere surrounding the hot surfaces, exhaust flames (and sometimes induction system flames) remain as a potential ignition source. According to previous research (reference 6), it has been established that effective induction system and exhaust system inerting could be accomplished by injecting CO_2 into the induction system to dilute the entering fuel-air mixture to the point of nonflammability.

It was decided for the AvSER full-scale crash test to close the butterfly throttle-valve in the carburetor at impact, thus allowing only "idle" air into the engine. At the same time, CO_2 would be injected into the compressor stage of the engine at an average rate of 0.5 pound per second. (See Experimental Procedure, CO_2 System, page 28). Closing the butterfly throttle-valve also lessens the possibility of ejection and loss of CO_2 by reverse flow through the carburetor. This valve was closed by CO_2 pressure acting through a series of mechanical linkages. The CO_2 bottle was mounted on the top rear of the carburetor in a relatively protected area. The mechanical linkage on the side of the carburetor, of which the throttle-valve closing mechanism was a part, was also in a relatively protected area. Figure 7 shows a side view of the mechanism after the actual crash.

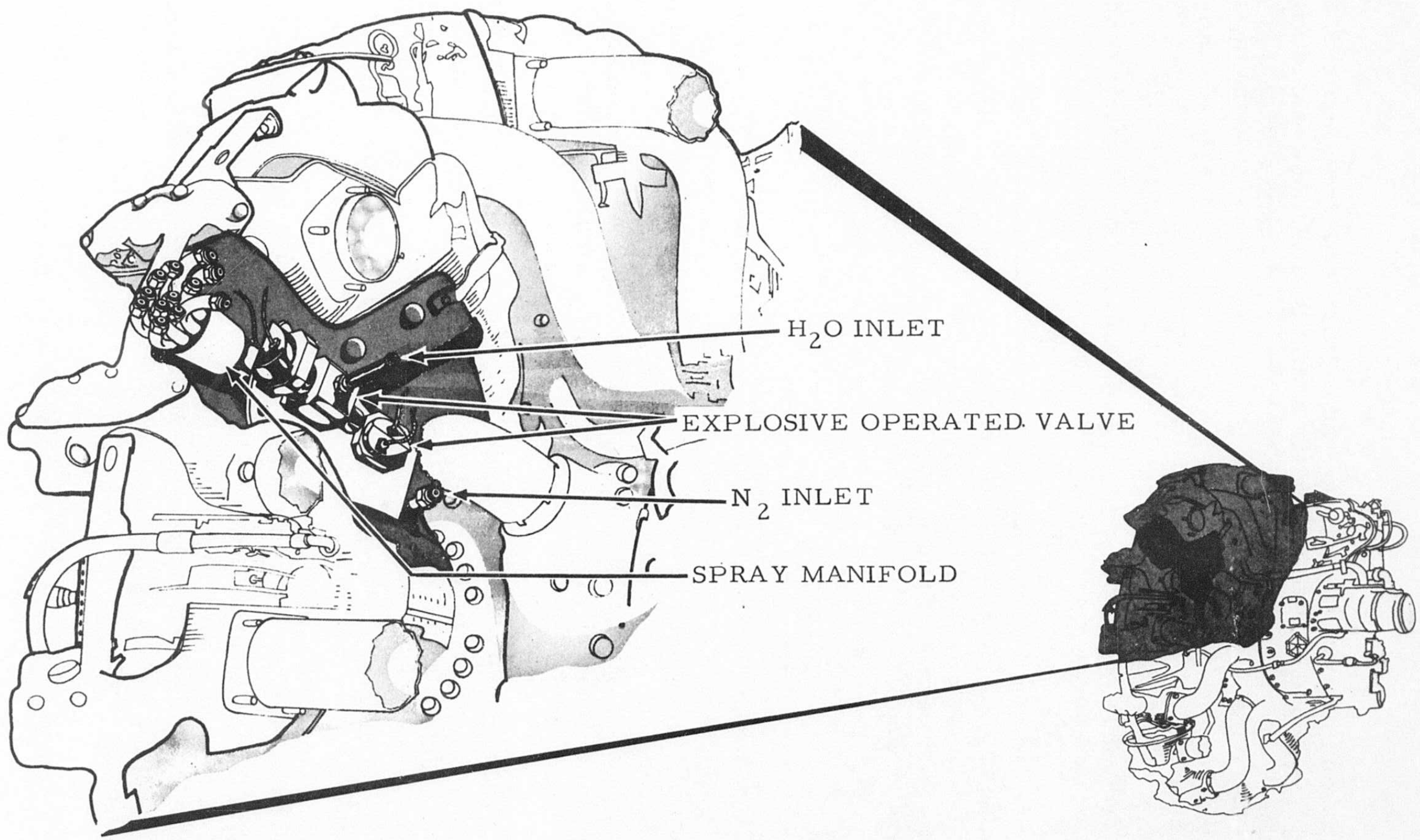

Figure 6. Orientation of Spray Distribution Manifold on the R1820-103 Engine. Two such units were used, one for the left-hand manifold and a second for the right-hand manifold.

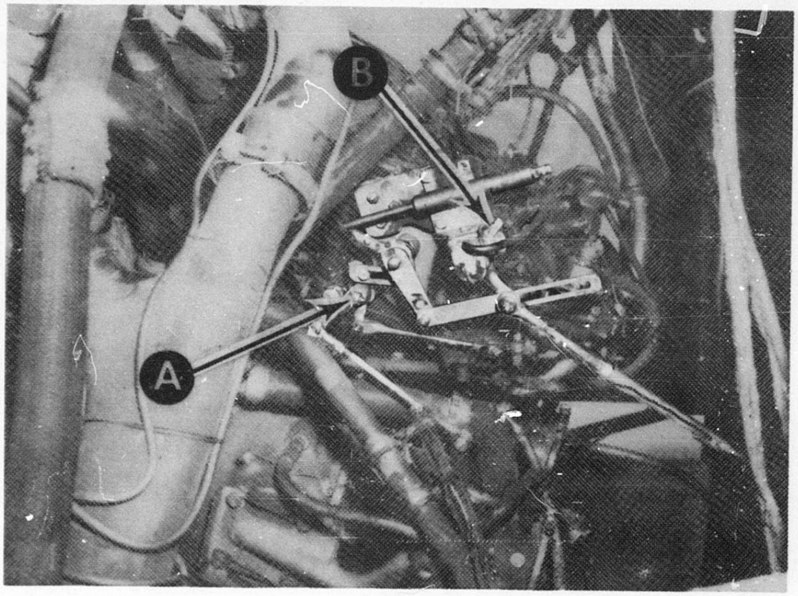

Figure 7. Carburetor Butterfly and Fuel Shutoff System. This is a postcrash photograph of the left side of the carburetor. The butterfly valve shaft (A) and the fuel cutoff (B) are visible. The CO_2 bottle, bracket and pressure lines have been removed for the photograph.

FUEL-OIL SHUTOFF SYSTEM

Shutting-off of the fuel supply at the carburetor was necessary for the internal inerting of the engine. This was accomplished by moving the carburetor mixture control to idle cutoff through a mechanical linkage actuated by CO_2 pressure after the explosive valve on the CO_2 system was fired. A schematic drawing of this system is shown in Figure 8.

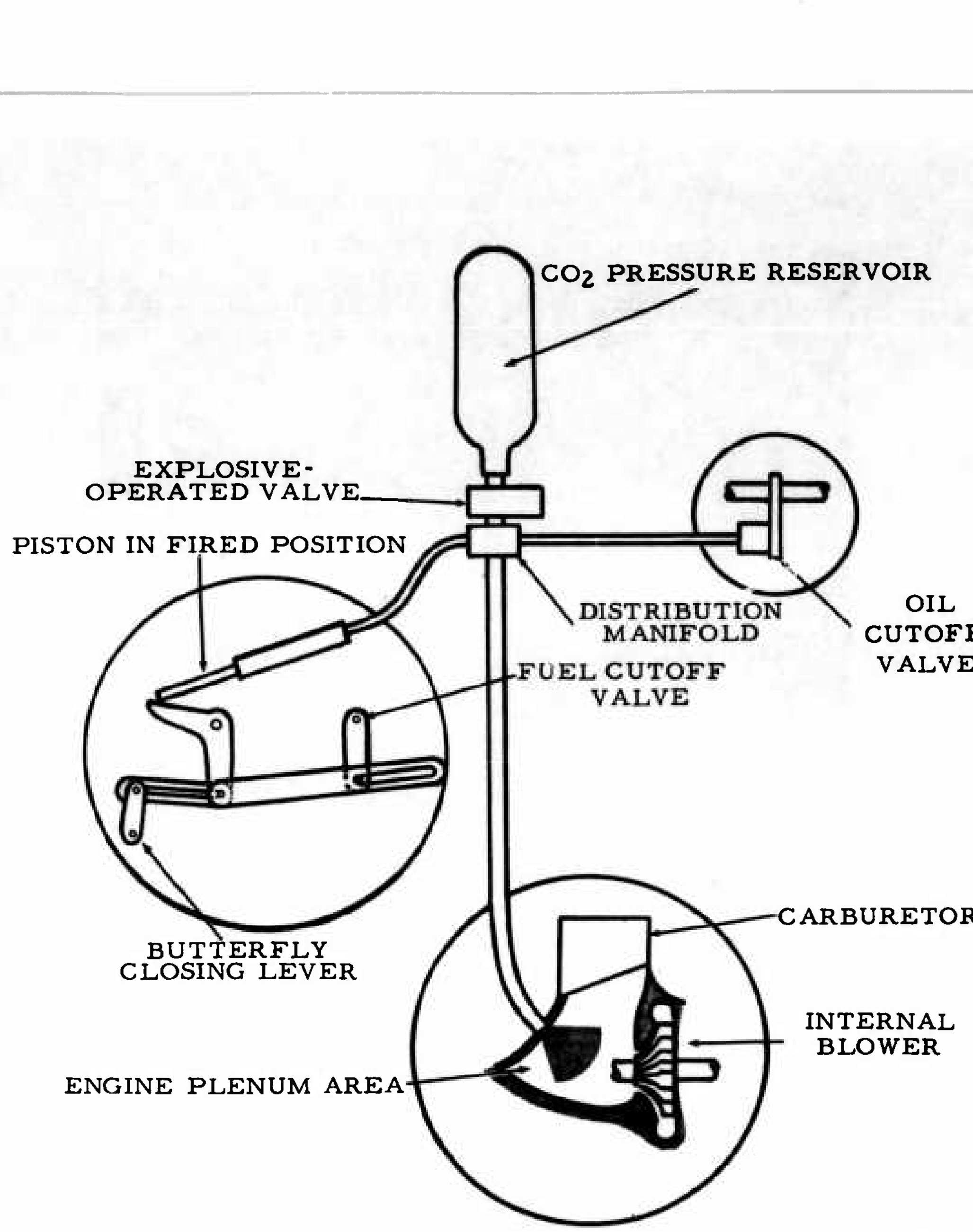

Figure 8. CO_2 System Schematic.

The oil was shut off at the point of its highest temperature, the engine outlet, prior to entering the oil cooler. The oil cutoff valve (Figure 9) was a modified Whitaker sliding-gate valve. The slow, electrically operated motor actuator normally used on this engine installation was replaced with a pneumatic rotary actuator having approximately 90 degrees of travel. Carbon dioxide was piped to the valve so that, when

this system was triggered, the pressure rotated the actuator, thus closing the sliding-gate valve. To reduce the possibility of damage during impact, the valve assembly was mounted near the engine in a protected area.

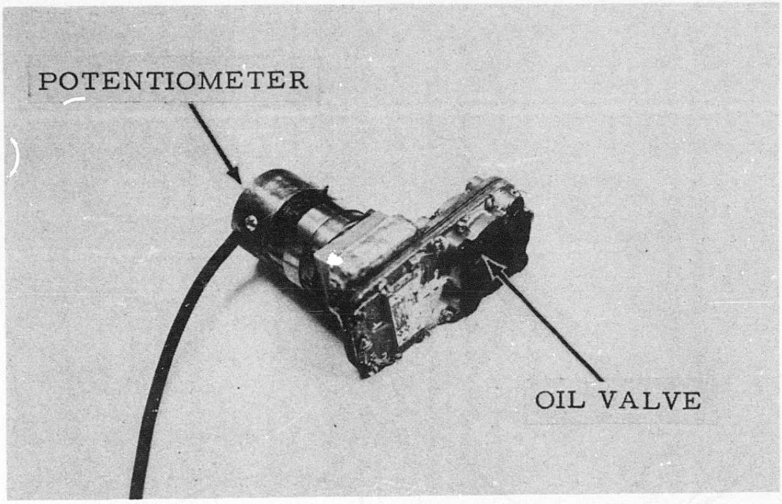

Figure 9. Oil Cutoff Valve.
A rotary pneumatic actuator replaced the electric motor on a standard Whitaker sliding-gate valve. A potentiometer was mounted on the valve to allow the recording of the valve position.

HYDRAULIC AND TRANSMISSION LUBRICATION SYSTEM

In the H-21 helicopter, there are three auxiliary systems using flammable fluids: one for the brakes (hydraulic), one for the flight controls (hydraulic), and one for lubrication of the transmission (oil). All of these systems involve small quantities of fluid and are in reasonably protected areas; thus, they provide a relatively small fire potential. No modifications to these systems were made.

ELECTRICAL SYSTEM

As previously stated, the primary objective of this study was to determine the feasibility of inerting the engine (internally and externally) within the time limit of 0.20 second after impact. The inerting of the

electrical system was not stressed in this experiment. However, to demonstrate the feasibility of inerting the electrical system within the 0. 20-second time limit, the following measures were taken. To restrain the battery more effectively, the battery tiedown was reinforced. The immediate area surrounding the battery was filled with a plastic foam to absorb some of the crash shock, to help insulate the area, and to provide better restraint. (See Figure 10.) Insulated explosive cable cutters were installed on the hot side of the battery and at the generator electrical outlet. In an installation on service aircraft, a disconnect capable of being reset while in flight would probably be used instead of the cable cutter.

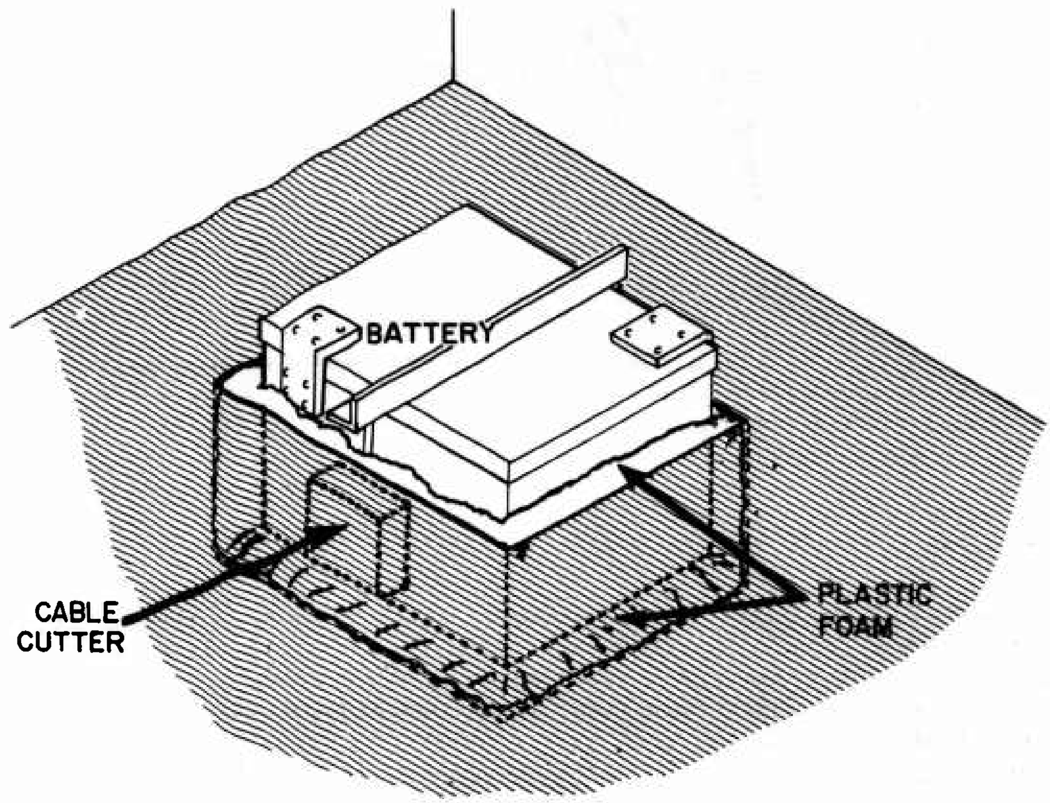

Figure 10. Battery-Box Area Modification.
An explosive cable cutter was mounted directly to the battery box. The immediate space surrounding the box was filled with foam plastic to help absorb the shock and to aid in insulating the area, and the two tiedowns were strengthened considerably.

Upon activation, the cable cutters were arranged to sever the main battery and generator cables in order to isolate the battery and generator. Because the power output from the inverters was not needed for the test flight, the inverters were left in the "off" position. (See Experimental Procedure - Electrical De-energizing, page 28, and Appendix I.) Figure 11 is a picture of the cable cutter.

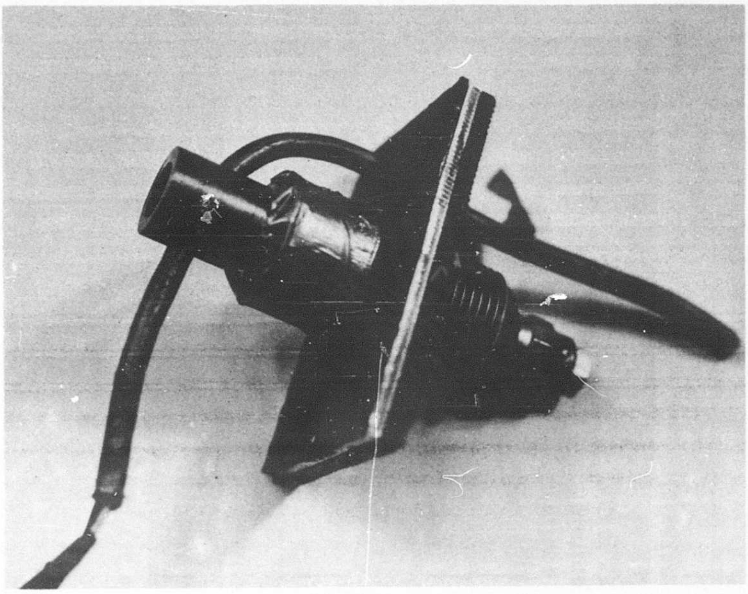

Figure 11. Explosive-Operated Cable Cutter.

INITIATION SYSTEM

The initiation system employed by NACA (references 6, 7, and 9) included rubber-strip-type deformation switches, cable-type deformation switches, skin deformation switches, and engine reaction switches. This system was so designed that several switches had to be activated to trigger the inerting system. In principle, such multiple switches provide a safety margin for the flying personnel.

For the crash test described here, three simple multi-directional impact switches and three contact switches were used. The three impact switches were mounted on the landing gear, and the three contact switches were installed on the bottom of the fuselage. Any one of the six switches was capable of triggering the system. All six were closed on, or shortly after, first contact of the aircraft with the ground. This initiation system cannot be considered adequate for operational aircraft. Further study of initiation systems will be required to develop a simple, reliable, and operationally safe method of triggering the inerting system.

EXPERIMENTAL PROCEDURE - PRELIMINARY TESTING

TIMING OF THE FIRE-INERTING SYSTEM

In determining how fast the entire system had to be in operation to prevent ignition, AvSER considered the previously reported fixed-wing crash-fire data. The time between impact and ignition reported in the NACA studies (reference 9) is shown in the following table for several crash fires conducted on the NACA program.

Ignition Source	Time in Seconds After Impact At Which Ignitions Were Observed
Hot Surfaces (Exhaust System)	1.3, 0.7, 1.9, 3.8
Exhaust Flames	4.1, 1.3, 2.0, 3.5, 1.9
Electrical Systems	
Arcs	1.0
Bulb Filaments	0.6
Induction Flames	2.2, 3.5, 7.7
Electrostatic Sparks	2.4
Chemical Agents	4.4, 3.5, 3.5

For an H-21 helicopter crashed by AvSER, a fire was observed starting to propagate through the fuel vapors approximately 0.58 second after impact with the ground. In earlier AvSER helicopter drop tests (water in the tanks in place of fuel), it was observed that fuel tank ruptures had taken place and massive "fuel" spillage was in progress by 0.20 second after impact.

After analyzing NACA's fixed-wing data, as well as AvSER's helicopter data, it was decided that for best protection against ignition, a fire inerting system for helicopters should be in operation in 0.20 second after first contact of the aircraft with the ground.

The length of time the spray system was required to remain in operation for inerting and cooling the engine exhaust ducting was determined experimentally. It was discovered that if the surface temperature of the exhaust collector ring was rapidly brought down to 400 degrees F. and the spray system was immediately turned off, the external surface of the exhaust collector ring started to reheat through conduction from

the hotter inner portions. By progressive tests, it was determined that a N_2-H_2O spray application for 27 to 30 seconds could lower the external surface temperature of the exhaust collector ring to 180 to 220 degrees F., and that at the termination of the spray the reheat process would raise the temperature of the exhaust collector ring to 380 degrees F. at the most critical points of the system.

DEVELOPMENT OF THE NITROGEN-WATER SPRAY SYSTEM

The water spray system requirements were obtained directly from a flyable H-21 helicopter. The test equipment was mounted on a 4- by 7-foot access panel, attached to the left side of the fuselage in the main landing gear area, near the plenum chamber. See Figure 12.

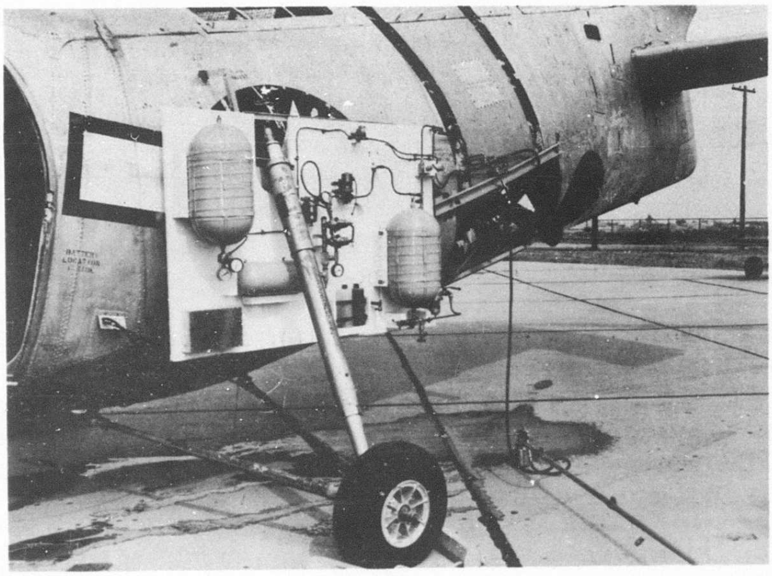

Figure 12. H-21 Test Bed.
This photograph shows the access board mounted on the side of the flyable helicopter. The equipment pictured above was used to develop a workable nitrogen-water spray system.

Items mounted on the access board consisted of a water reservoir, a high-pressure nitrogen reservoir, pressure regulators, needle valves for N_2 and H_2O flow rate adjustment, filters, and solenoid valves.* It also held an additional low pressure air supply (for injecting the fuel-air mixtures in the "time to suppress tests" discussed below) and the necessary wiring circuits for measuring, with thermocouples, the actual exhaust manifold cooling rates during the tests.

Figures 13 and 14 are typical of many such curves obtained with various orifice combinations. From these data it was determined that the most effective orifice sizes were 0.055 inch for N_2 and 0.082-inch for H_2O. The actual quantity of N_2 and H_2O needed was also established in these tests.

The effectiveness of the inert atmosphere surrounding the exhaust collector ring was determined by "fire suppression" tests. While the engine was running at maximum operating temperature, raw fuels were sprayed into the space between the shroud and the exhaust collector ring. The spray was controlled by the regulated, low-pressure air in a movable spray gun. When a definite fire was established, the nitrogen-water system was activated and the time between activation of the inerting system and fire suppression was observed. Figure 15 is a photograph of the test panel in operation. A fire is visible extending out of the shroud on top of the extension attached to the exhaust collector ring.

By using an exhaust extension, (Figure 16), normal engine torching and the test fires were separated. The flammable fluid spray gun seen in Figure 16 was moved from one area to another while testing the inerting effect of the nitrogen-water spray system. Also visible in this figure is the spark-plug arrangement set up to insure immediate ignition of the flammable fluids.

While it was definitely established that, with the engine running, a fire could be ignited by the hot exhaust collector ring, it was at times difficult to get the proper mixture. The high-speed air flow between the exhaust collector ring and the shroud helped to act as a heat dissipater (reference 15), thus making it more difficult to ignite the fuel at the higher engine rpm. To insure a definite ignition source for the flammable fluids for the tests, a spark-plug and magneto system was used.

* Solenoid valves were used in the early testing for convenience in place of explosive valves. In timing the operation of the system from initiation to the desired cooling temperature, the operating time for the solenoid valve action was taken into consideration.

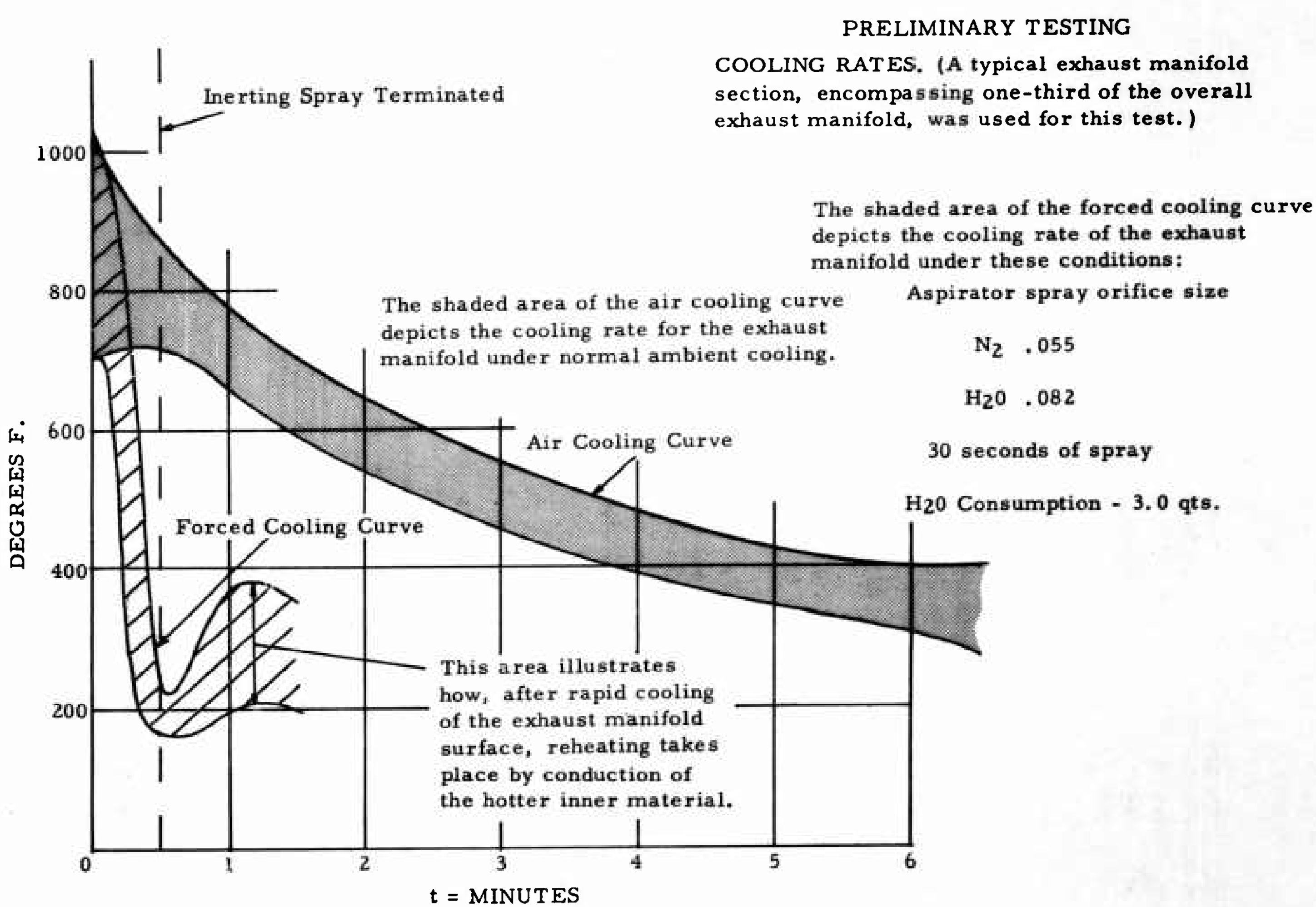

Figure 13. Manifold Cooling Rates for N_2-H_2O Orifices 0.055/0.082.

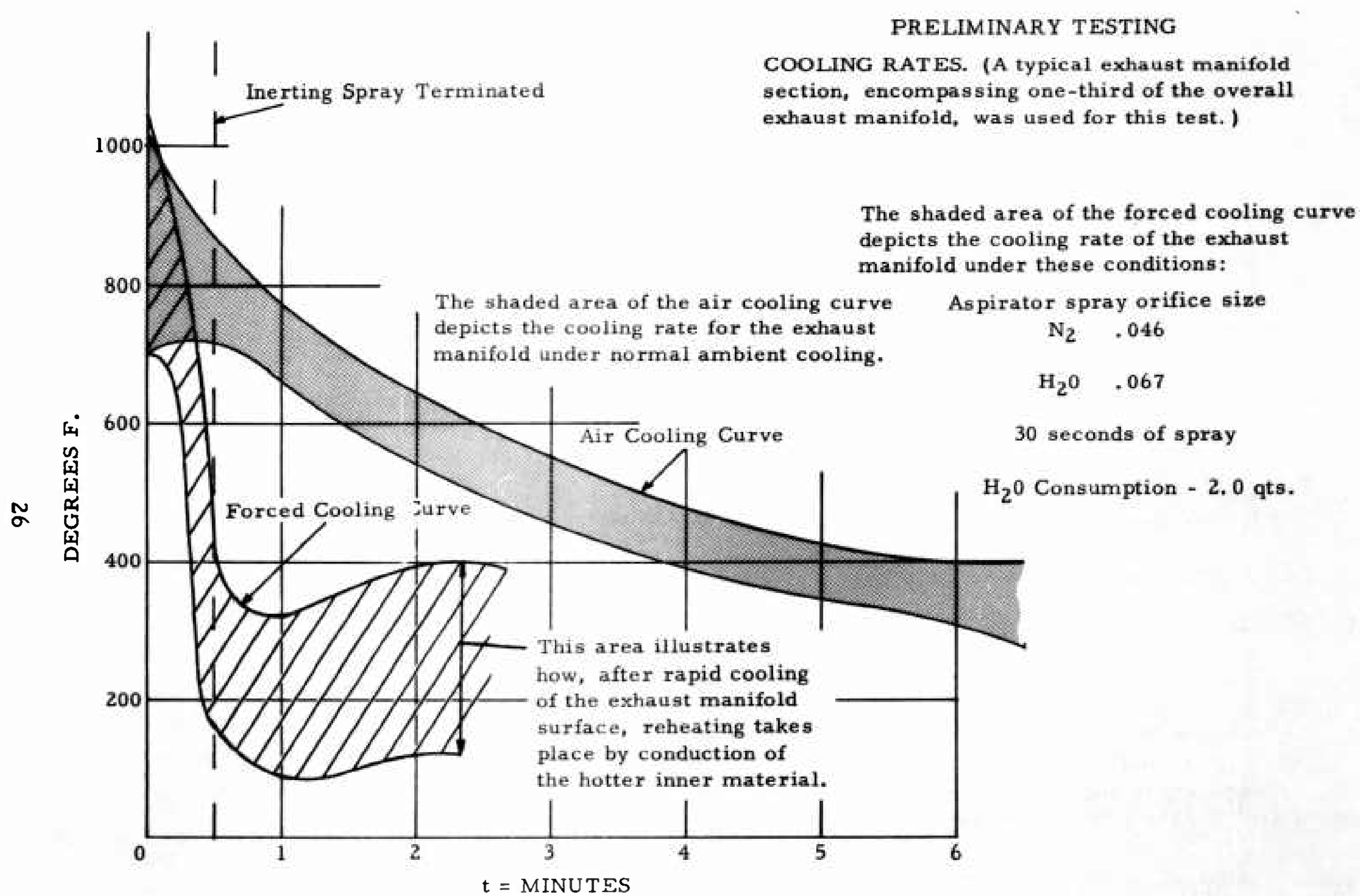

Figure 14. Manifold Cooling Rates for N_2-H_2O Orifices 0.046/0.067.

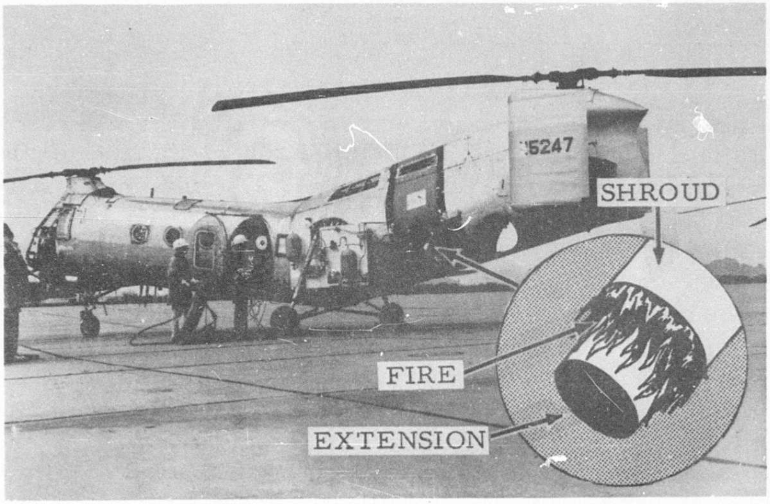

Figure 15. Fire Suppression Test.
Fire is visible in the space between the exhaust extension and the shroud, during the early "time to suppress" fire tests.

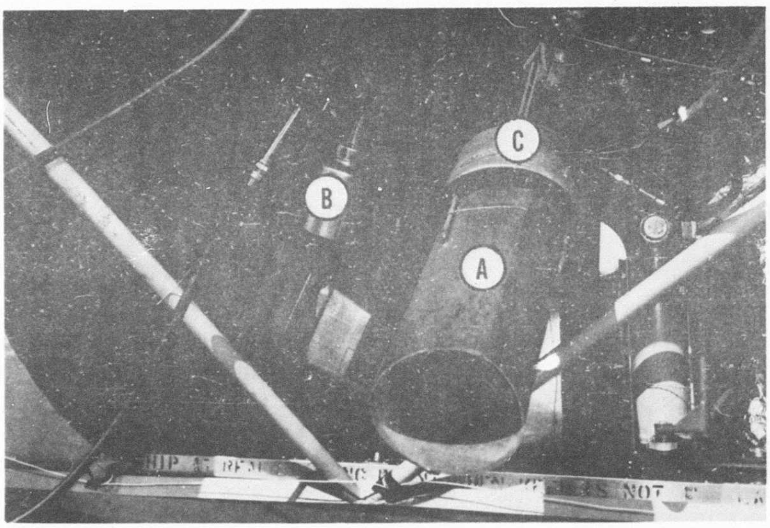

Figure 16. Exhaust Extension.
The protruding pipe (A) is an extension of the exhaust collector ring. The movable spray gun (B) was used for spraying flammable fluids into the area between the exhaust collector ring and the shroud. A spark plug (C) was used to provide positive ignition in some of the fire suppression tests.

DEVELOPMENT OF THE CO_2 SYSTEM

In designing the CO_2 system, several factors were considered, including (a) the rate of CO_2 discharge required to inert the interior of the engine, (b) the duration of the CO_2 being discharged, and (c) the effect of engine-compartment temperature on the CO_2 system pressure.

It can be shown that the following conditions exist upon closing the butterfly throttle-valve and injecting the CO_2 into the compressor section of the R-1820-103 engine.

1. The air flow into the engine is reduced from approximately 1.4 pounds/second, (assuming 2500 rpm and 40 inches of Hg manifold pressure) to approximately 0.35 pound /second, assuming 2500 rpm and 10 inches of Hg manifold pressure (absolute).

2. The injected CO_2 (0.50 pound/second will dilute the incoming air-fuel mixture to the point of nonflammability.

In tests measuring time from fuel cutoff until engine stoppage, it was established that, depending upon the engine rpm at the start of engine shut-down, the engine coasted for 3 to 6 seconds with the blades disengaged. For this reason, it was decided to provide the CO_2 injection for at least 8 seconds, thus insuring CO_2 flow during the entire engine stopping period.

In determining the size of the CO_2 reservoir, one must consider the engine compartment operating temperature, since an increase in temperature results in rapid pressure buildup within the CO_2 reservoir. From tests, it was determined that the temperature in the engine compartment varied from 120 to 150 degrees F. during normal operation. A standard 200-cubic-inch CO_2 reservoir filled with 4-1/2 pounds of CO_2 (58 percent full), which would provide an adequate safety margin at 180 degrees F., was selected for use in the AvSER test. Since it is desirable to have the CO_2 bottle as close as possible to the discharge points, the bottle was installed on the top rear of the carburetor.

DEVELOPMENT OF THE ELECTRICAL DE-ENERGIZING SYSTEM

Electrical ignition of the flammable fluids in aircraft accidents may occur from:

1. Sparks or arcs resulting from short circuits in wiring.

2. Sparks or arcs from starters, magnetos, generators, batteries, and inverters.

3. Spark discharges from the brushes of rotating equipment.

4. Electric arcs created by the separation of contacts in an electric circuit, such as in a voltage regulator.

In designing an electrical de-energizing system, the primary items to be inerted are the battery, the generator, the magnetos, and the inverter. All but the magnetos present no special problem. The magnetos, however, present a point of special interest. A spark of an intensity to ignite the fuel is emitted 18 times per 2 engine revolutions (2 magnetos), with separate wires running to 18 spark plugs. A potential fire ignition source is obviously established. However, to de-energize the magnetos at the time of impact would result in not igniting the fuel being inducted into the engine. Any raw fuel in the engine after impact would be forced out through the exhaust system. Since the exhaust system is considerably hotter than the ignition temperature of the mixture, an exhaust system fire would frequently occur, with good possibility of igniting fuel spilled from various sources during impact.

According to references 6 and 9, the safest way to insure against an induction or exhaust fire is to leave the ignition system on during the crash sequence. Fortunately, in the H-21 helicopter, the spark ignition wires are shielded fairly well and are located in an area that, judging from H-21 accident experience, remains reasonably intact. Therefore, it was decided to leave the ignition system in operation during the crash.

By removing the battery from the aircraft circuit and at the same time eliminating the output of the generator and the inverter, the electrical system is eliminated as a potential ignition source. In this test, the generator was to be removed from the system by an insulated, explosive-actuated cable cutter installed on the generator output line. The battery was to be isolated likewise by cutting the "hot" side at the battery box. The explosive-operated cutter is shown in Figure 11.

For this crash test, it was decided that, since the inverter was not needed for actual flight, the inverter would be left in the "off" position. The inverter does provide a potential source of ignition (reference Appendix I), and for service aircraft a disconnect at the inverter must be provided.

It is anticipated that in future tests, fast-acting relays or devices providing a rapid recycle capability will be used to inert both the AC and DC power systems.

EXPERIMENTAL PROCEDURE - CRASH TEST

The intent of the crash, from the standpoint of the fire-inerting system,* was to test the operation of the system under crash conditions. A small external fuel tank was used to provide an additional safety margin by reducing the quantity of flammable fluids on board.

FUEL-TANK MODIFICATION

Fuel for the test flight was supplied from an 18-gallon tank mounted outside the aircraft on the left horizontal stabilizer. The fuel was gravity fed to the carburetor through a flexible line allowing for 2 feet of engine displacement. The aircraft main fuel tank was sealed off and filled with 200 gallons of water. Approximately 3-1/2 gallons of aviation gasoline were also left in the tank. The fuel was limited in this way so that if a postcrash fire occurred it would not destroy salvable test equipment needed for later tests and so that a better analysis of damage to the aircraft could be made.

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* The helicopter contained anthropomorphic dummies and experimental seats and litters. The fire-inerting system was only a small portion of the overall crash test. See TRECOM Technical reports on U. S. Army H-21 Helicopter Crash Test, 7 September 1962; TRECOM 63-3, <u>Dynamic Test of an Aircraft Litter Installation</u>, U. S. Army Transportation Research Command, Fort Eustis, Virginia, March 1963; TRECOM 63-24, <u>Dynamic Test of a Commercial-Type Passenger Seat Installation in an H-21 Helicopter</u>, U. S. Army Transportation Research Command, Fort Eustis, Virginia, June 1963; TRECOM 63-62, <u>Dynamic Test of an Experimental Troop Seat Installation in an H-21 Helicopter</u>, U. S. Army Transportation Research Command, Fort Eustis, Virginia, November 1962; and <u>H-21 Helicopter Airframe Deformation Under a Dynamic Crash Condition</u>, report to be published.

INSTRUMENTATION

A list of the instrumentation related to this experiment is presented in the following table.

INSTRUMENTATION LIST

Device	Provided	Location	Specification
High-Speed Motion Picture Camera	Displacement/time for helicopter	4 on ground 1 on aircraft	Photosonics 1B high G tolerance, 500 fps 16mm Ektachrome ER 430
Normal-Speed Motion Picture Camera	General photographic coverage	4 on ground	2 Kodak 16mm 64 fps, 2 Bolex 16 mm 24 fps, Kodachrome II
Pressure Transducers	Operational Timing of $N_2 + CO_2$ systems	1 N_2 system 1 CO_2 system	0 - 2000-psi pressure transducer
Potentiometer	Operational timing of fuel/oil cutoff	1 fuel cutoff 1 oil cutoff	Potentiometer transducer
Recording Oscillograph	Amplitude-time records of transducer outputs	4 each at ground control point	CEC model 5-114,26 channel recording oscillograph with related power supplies
Photograph-Oscillograph Data Correlation Device	Zero time for camera film and oscillograph record	2 each	Photo flash bulbs mounted in field of view of cameras Firing pulse to bulbs recorded on oscillograph record for correlation.
Voltage Generator	Timing for high-speed cameras	Ground control point	120-volt AC generator, 60 cps timing in cameras. Also recorded on oscillograph records.

TEST CRASH

The test flight, conducted by radio control, resulted in the flight path shown in Figure 25I, Appendix III. The conditions at impact are shown in Figure 17.

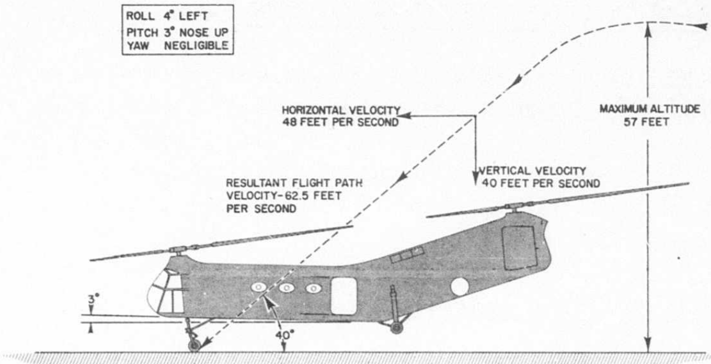


Figure 17. Velocity and Attitude at Impact.

As the aircraft struck the ground and skidded forward, it yawed to the right and came to rest at approximately 45 degrees from the line of flight and in the upright position as shown in Figure 18. The sequence of photographs (Figure 25) presented in Appendix III illustrates the kinematic behavior of the aircraft during the first 3 seconds following impact. At 1.46 seconds after nose-wheel impact, a fire involving the 3-1/2 gallons of aviation gasoline in the main fuel tank appeared. The possible ignition sources are discussed in the following section.

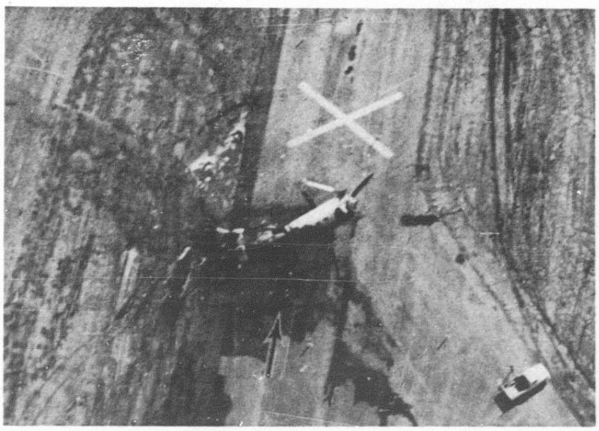


Figure 18. Postcrash View.
This photograph illustrates the final resting position of the helicopter. The forward portion of the helicopter is yawed approximately 45 degrees from the line of flight (arrow) and the aft section is yawed an additional 15 degrees.

EXPERIMENTAL RESULTS

An on-the-spot postcrash investigation revealed the following:

OVERALL AFT SECTION OF HELICOPTER

The aft section remained relatively intact during the crash. The fire, which was first visible on the bottom near the main landing gear V-brace attachment, enveloped the tail section by propagation along the ground. Because the fuel load had been limited, the fire damage was confined generally to wire insulation, rubberized canvas baffling, and miscellaneous thin aluminum tubes.

ENGINE COMPARTMENT AREA

1. The aluminum hollow push-pull throttle-linkage tube (rod assembly, throttle, 22C2338) which runs between the cross-beam-assembly bell crank and the engine-supported bell crank was burned in two.

2. The aluminum mixture control rod that runs parallel to the throttle linkage was reduced to approximately 50 percent of its original cross section. (See Figure 19.) While the magnitude and duration of the engine compartment fire were relatively small, the seriousness of the situation is clearly illustrated. Of the two engine control linkages used by the pilot, one was rendered useless and the other was severely damaged. Should this have occurred in an in-flight fire, primary engine control would have been lost. The use of steel linkages for such control rods would be preferable.

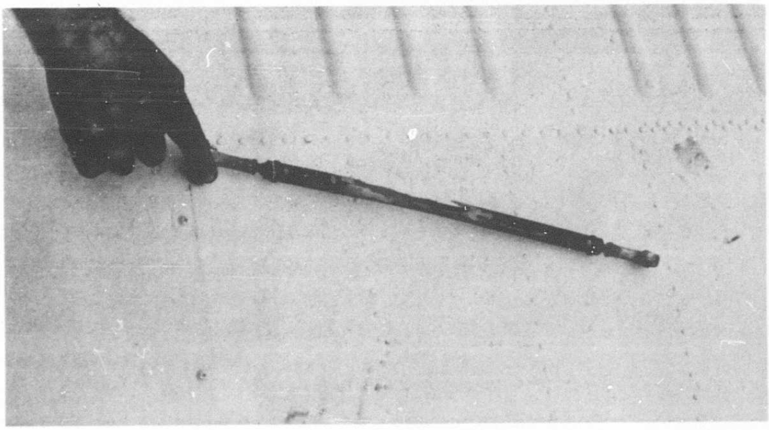


Figure 19. Mixture Control Rod.

3. No structural failures occurred in the fuselage motor-mount attachments.

4. The engine motor-mount assemblies (Lord Mounts) remained intact.

5. The generator cable cutter fired, initiating the cutter timing pulse, but failed to sever the cable due to an inadequate powder charge.

6. The battery cable cutter fired, initiating the cutter timing pulse, but failed to sever the cable due to an inadequate powder charge.

7. The mounting assembly for the N_2-H_2O system and regulators held satisfactorily, with only a minimum of deformation resulting.

8. The mounting assembly for the spray manifold, N_2-H_2O explosive-operated valves, and the aspirator remained undamaged.

9. The CO_2 system mounting brackets held satisfactorily.

10. The entire spray distribution tube network remained intact and undamaged.

11. All of the explosive charges, two per valve (10 total) and one per cable cutter (2 total), fired.

SEQUENCE OF INERTING OPERATIONS

The sequence in which certain events occurred during the crash is shown in Figure 20. (See Appendix V.) It will be observed that:

1. The first contact at time zero was at the nose wheel.

2. The main gear contacted the ground at 0.052 second.

3. The fire-inerting system was triggered 0.069 second after nose wheel contact.

4. The battery and generator cable cutters fired at 0.074 second.

5. By 0.075 second after impact a definite flow occurred in the N_2 and CO_2 systems.

6. By 0.090 second the fuel supply was cut off at the carburetor.

7. By 0.120 second after impact the oil line was closed at the engine.

8. At 0.124 second, the aircraft fuselage underbelly impacted the ground.

9. The first indication of cooling took place at 0.200 second after nose wheel impact.

10. The first visible indication of fuel spillage occurred at 0.370 second.

11. The first visible fire appeared at 1.462 seconds.

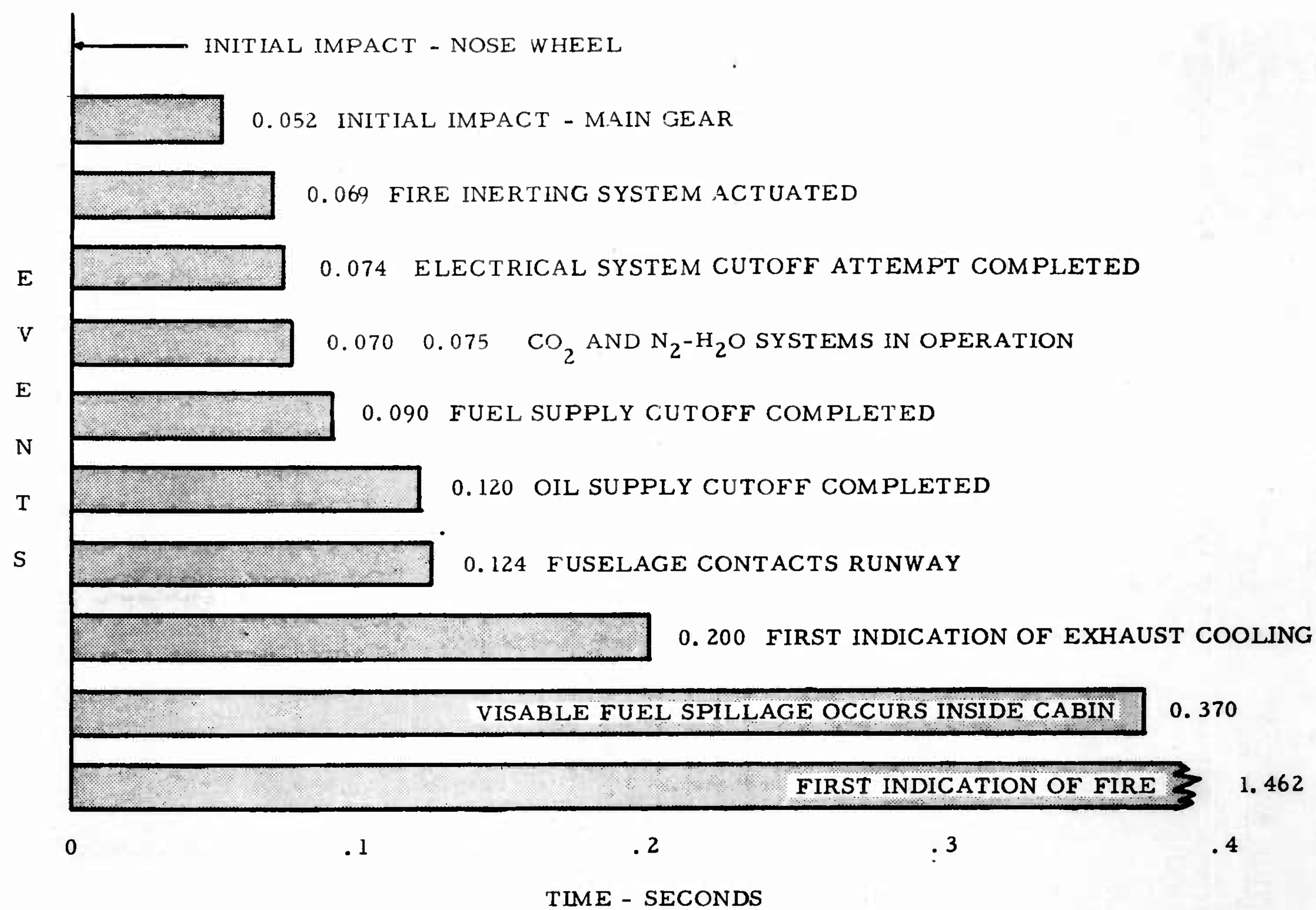


Figure 20. Event Timing.

EVALUATION

INITIATION SYSTEM

The initiation system functioned as designed with all inertia switches and all contact switches (total of 6 switches) being closed.

The battery for the initiation system (located in the plenum chamber) survived the crash satisfactorily; however, the wires leading from the battery to the various explosive charges were burned, with evidence of shorting. Upon examination, it was discovered that the burned wires were pinched during breakup of the structure. Since the battery is no longer needed after the fuselage strikes the ground, it should have been de-energized after it had triggered the system. A more refined system being planned for future tests will accomplish this. Armor and other safety devices can be provided for those portions of the initiation system which cannot be immediately de-energized.

NITROGEN-WATER SPRAY SYSTEM

The nitrogen-water system apparently functioned satisfactorily. The explosive-operated valves fired at 0.069 second after nose-wheel contact; by the time the fuselage contacted the runway (.124 second from nose-wheel contact), the entire spray system was in operation. Thermocouples on the exhaust ducting gave first indication of spray cooling at 0.20 second after impact. Because three of the six thermocouples were dislodged during the crash, an accurate evaluation of the overall cooling effectiveness was not possible. However, since three of the thermocouples gave valid readings and since the effectiveness of the spray cooling system was verified during the numerous static tests (Figure 13), it can be assumed that satisfactory cooling was achieved.

A postcrash check of the equipment revealed that the H_2O reservoir accumulator, the N_2 pressure vessel, the regulators, and the low-pressure manifold were undamaged. The engine-mounted explosive valves, the spray manifold, the aspirator, and the entire spray distribution tube network also survived the crash with no damage.

CO_2 SYSTEM

The CO_2 system functioned as designed. In 0.070 to 0.075 second after impact, the CO_2 system was starting to discharge into the carburetor. By 0.09 second, the gas pressure had closed the carburetor butterfly valve and had moved the mixture linkage in the carburetor to idle cutoff.

By 0.12 second, the gas pressure had fully closed the oil cutoff valve.

The effectiveness of the CO_2 injection into the carburetor is best evaluated by the fact that no flame is apparent at the exhaust outlet in any of the high-speed pictures.

ELECTRICAL INERTING SYSTEM

The electrical inerting system did not function as designed. Though sections of cable were satisfactorily severed in static tests, both the battery and generator cables were not completely severed in the final crash flight. However, it should be pointed out here, that in a final inerting system, fast-acting relays would probably be used in place of cable cutters. This would provide a re-cycle capability.

INSTRUMENTATION

The pressure transducers used on the N_2 and CO_2 systems for indicating that these systems were turned on, operated satisfactorily. However, by installing the pressure transducers downstream of the "on-off" explosive valves, a better initial pressure surge indication would result. The two potentiometers used to record movement of the fuel and oil cutoff functioned satisfactorily.

FIRE

The main fuel tank contained 200 gallons of water and approximately 3-1/2 gallons of aviation gasoline. It can be assumed that this fuel was lying on top of the 200 gallons of water at impact. Liquid was observed streaming from the fuselage mid-section 0.40 second after impact as shown in Figure 21. As the aircraft skidded forward and yawed to the right, a large "fuel" pattern developed directly under the aft portion of the helicopter. At 1.462 seconds after impact, ignition of the gasoline vapors occurred. Based upon high-speed film and post-crash investigation, the fire was ignited by one of two sources:

1. Abraded sparks from the steel landing gear V-brace attachment points on the belly of the fuselage. See Figure 22. From study of the high-speed films, ignition appeared to occur external to the fuselage, as seen in Figure 23.

2. Shorted wires running from the initiation system battery mounted in the plenum chamber. During the postcrash investigation, it was observed that the wires leading from this battery were pinched in two places between torn structure. The wire insulation appeared to be burned from the inside out, thus indicating a shorted wire.

ABRADED SPARK PROBLEM

A study has been made by NACA (reference 5) to determine whether common aircraft metals produce friction sparks capable of igniting combustibles that might be spilled in an airplane crash. Samples of aluminum, titanium, magnesium, chrome-molybdenum steel, and stainless steel were dragged over both concrete and asphalt runways while combustible mixtures of gasoline, JP-4 fuel, kerosene, and preheated oil were sprayed around the sample.

No ignitions occurred from aluminum at bearing pressures up to 1455 pounds per square inch and slide speeds up to 40 miles per hour. From a study of these results and of related literature, it is believed that aluminum would not be an ignition source even at higher bearing pressures and slide speeds. Titanium, magnesium, chrome-molybdenum steel, and stainless steel produced friction sparks that ignited the combustibles at sliding speeds and bearing pressures well below those that could be expected in a crash.

A solution to the spark ignition problem may be to clad dangerous metals with aluminum or other non sparking materials like beryllium-copper. Only those parts which may contact the ground in a crash need to be clad. The proper use of clading for this purpose needs further evaluation in realistic tests.

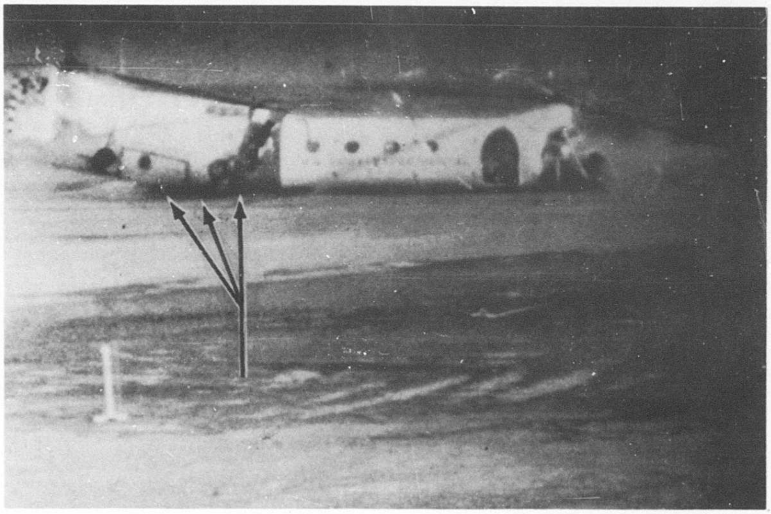


Figure 21. Fuel Flow at 0.40 Second After Impact.

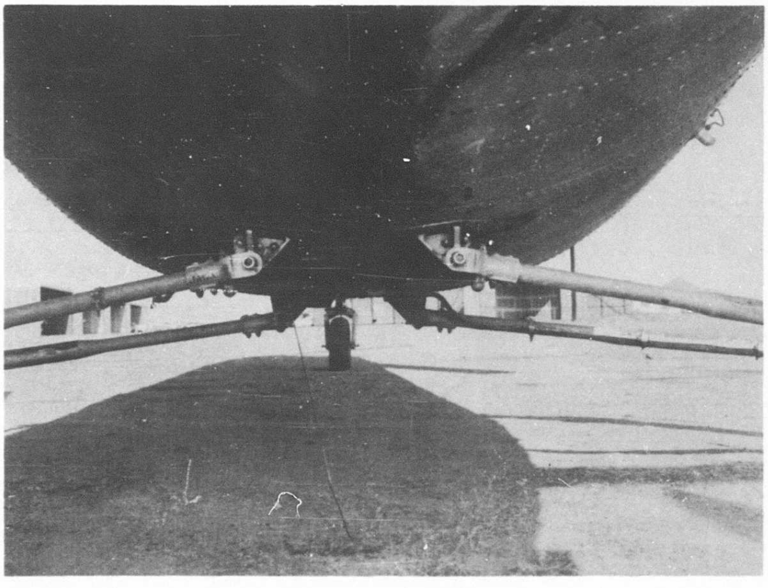
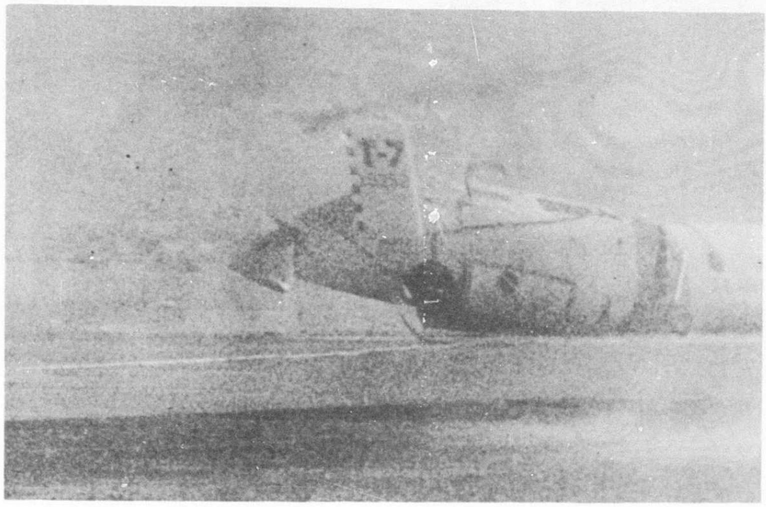
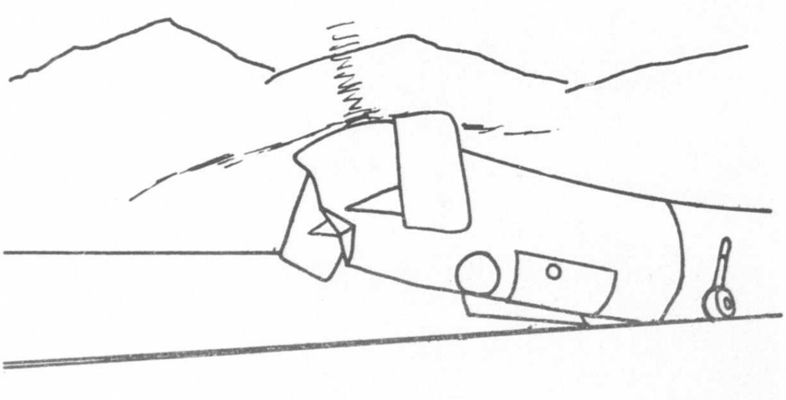


Figure 22. Lower Landing Gear V-Brace Attachments.

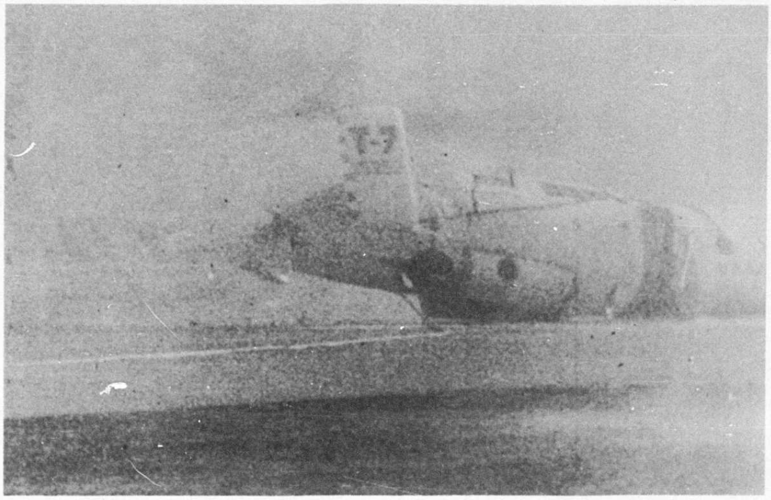
Figure 23. Sequence Photographs Showing Fire Initiation.



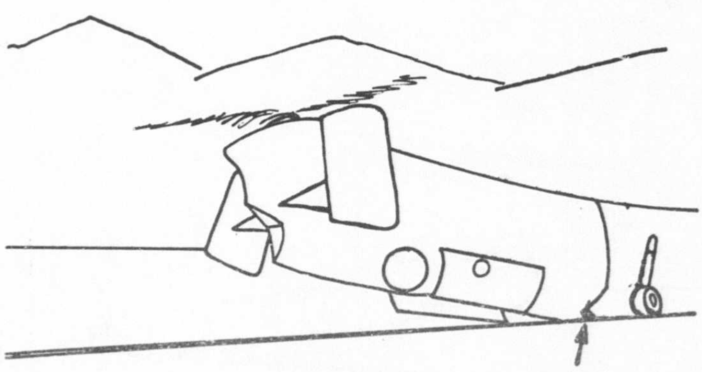
23A Series photograph taken from a 480-fps motion picture camera at 1.441 seconds after impact, no fire visible.



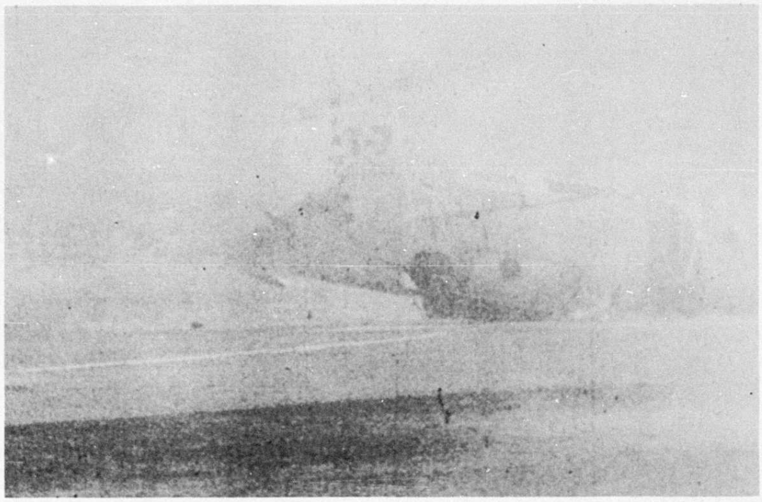
23B Same scene as above. Sketch is to help identify parts of the picture that are obscured during the 16mm enlargement and printing reproductions.



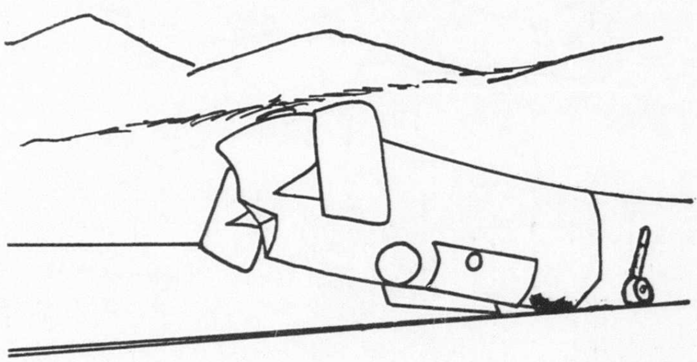
23C At 1.462 seconds after impact, fire is first visible along the bottom of the aircraft where the landing gear V-brace punctured the skin near the rear landing gear attachment point (see Figure 23B).



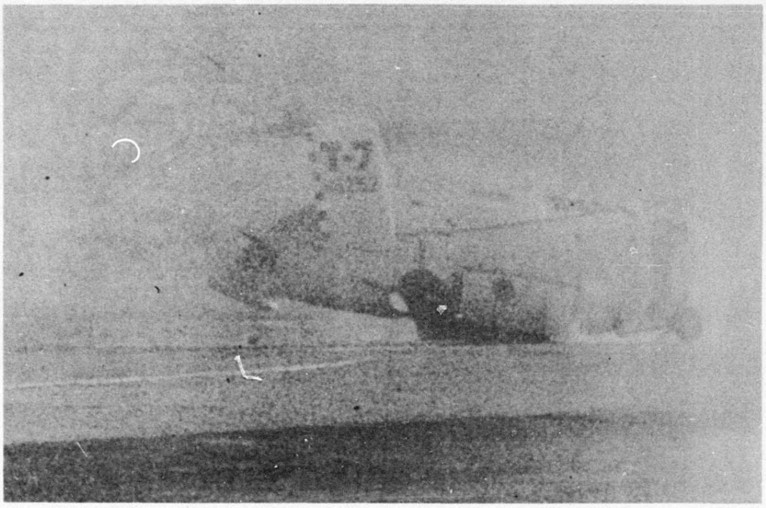
23D Schematic of photo above.



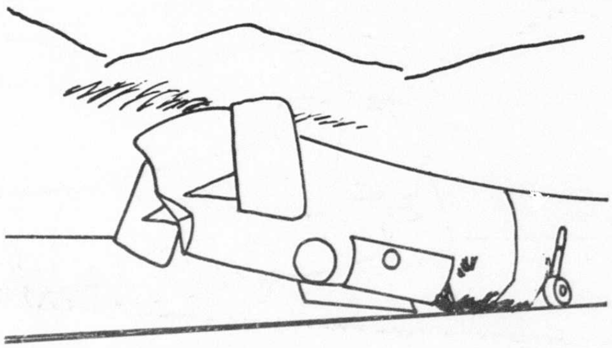
23E At 1.483 seconds after impact the flame is starting to propagate fore and aft along the ground surface.



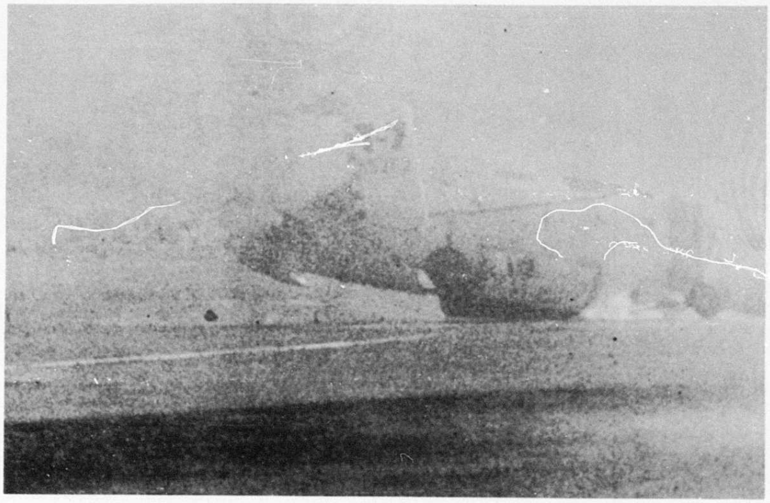
23F Schematic of above photograph.



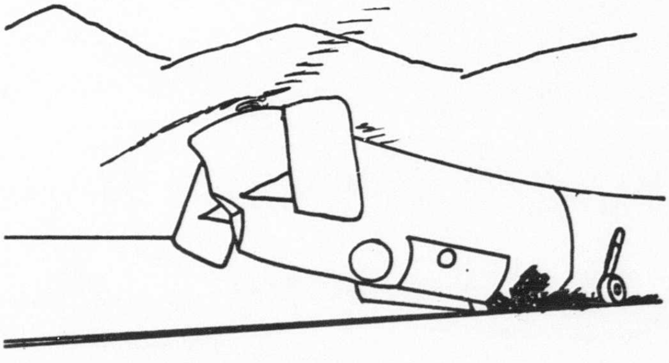
23G At 1.5136 seconds after impact the fire has propagated forward about 6 feet and aft to the wedged-open forward portion of the engine-plenum-chamber doors.



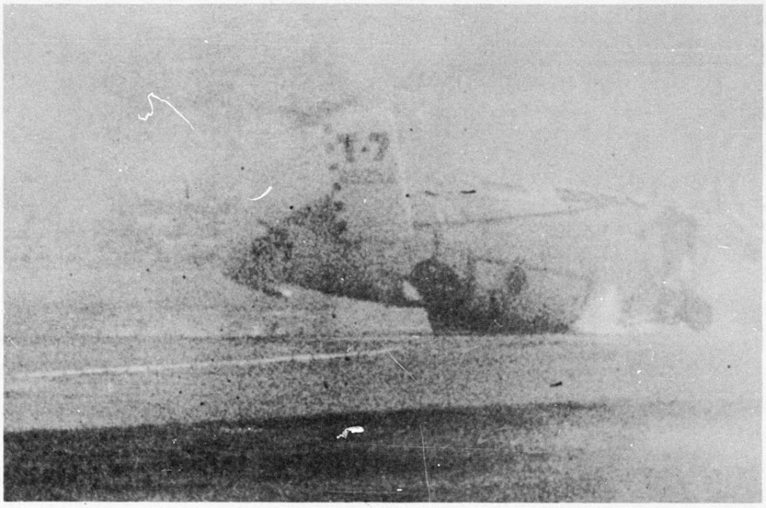
23H Schematic of photo above.



23I By 1.5444 seconds the fire has started to build. The vapors are on fire in the plenum chamber and in the area occupied by the fuel tank.

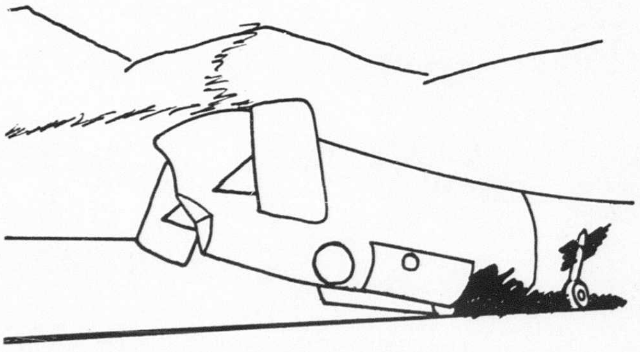


23J Schematic of above photo.



23K Burning fuel vapors in the forward portion of the plenum chamber and fuel tank area are starting to spout from the access doors and the skin rupture. Time: 1.5652 seconds after impact.

No flame has appeared in the engine compartment. As the aircraft continues to yaw to the right and the tail settles to the ground, the flame propagates along the surface of the spilled fuel and eventually engulfs the tail section.



23L Schematic of above photograph.

REFERENCES

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APPENDIX I. THE INVERTER AS A FIRE IGNITION SOURCE

Several tests were conducted with the H-21 inverter* to determine whether a fire could actually be started by the commutator sparks or by the 115-volt, 500-cycle alternating current being generated.

The inverter was installed on a test platform and wired so that it could be controlled from a remote station. The cooling fan was removed, primarily to eliminate the forced air flow through the inverter, thus creating a more favorable situation from the standpoint of ignition (references 9, 14, and 15). The AC output wires were arranged in such a way that they simulated an actual wire being severed, thus creating electrical sparks associated with the arcing of broken wires.

With the inverter in operation, aviation gasoline, hydraulic fluid, and lubricating oil were sprayed into and around the commutator area. No ignition occurred. The flammable fluids were also sprayed directly on open-air arcs produced between the AC output leads. In some 20 tests, no ignition took place.

In order to create more effectively a combustible zone or area, a cotton wick was dampened with the same flammable fluids. The wick was placed in the inverter next to the commutator sparks. The degree of wetting was varied in order to insure, at one time or another, conditions most suitable for ignition. No ignition could be detected.

The same wick arrangement was placed adjacent to the 400-cycle, 115-volt AC arc. Ignition readily took place.

From these tests, it was concluded that the likelihood of fire ignition from the commutator sparks is practically negligible; however, the 115-volt, 400-cycle alternating current does present a potential ignition source.

It was shown by actual test that the time between DC input cutoff to the inverter and AC output cutoff within the inverter was 0.385 second. Thus, based upon the desired time of 0.20 second for inerting, cutting of the DC power supply to the inverter would not accomplish the desired result. It is recommended that the AC output also be cutoff at the inverter outlet.

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* Inverter - Class A, 500 Volt Amp, 3 Phase, AN3533-1, input 27.5 VDC, 37 Amp. DC output, 3 Phase, 115 Volts, 400 Cycles. 500 Volt Amp. 0.90 Power Factor Mfg. Bendix.

APPENDIX II. IGNITION BY HEATED SURFACES

Hot engine metal is a potential ignition source for the various flammable fluids carried aboard aircraft. In determining the temperature to which the hot surfaces must be cooled in order to prevent ignition, a thorough review of the available literature was made.

While there are some differences of opinion as to what this temperature should be, there remain several conditions that are mutually agreed upon. These are:

1. As the temperature of the combustible fluid increases, the rate at which its vapors are given off increases. Thus, if ignition is to take place, it will do so more quickly with a hot fluid, since a combustible mixture will be reached earlier.

2. A relation between surface temperature and contact time required for ignition is given in Figure 24 for JP-4 fuel (reference 8).

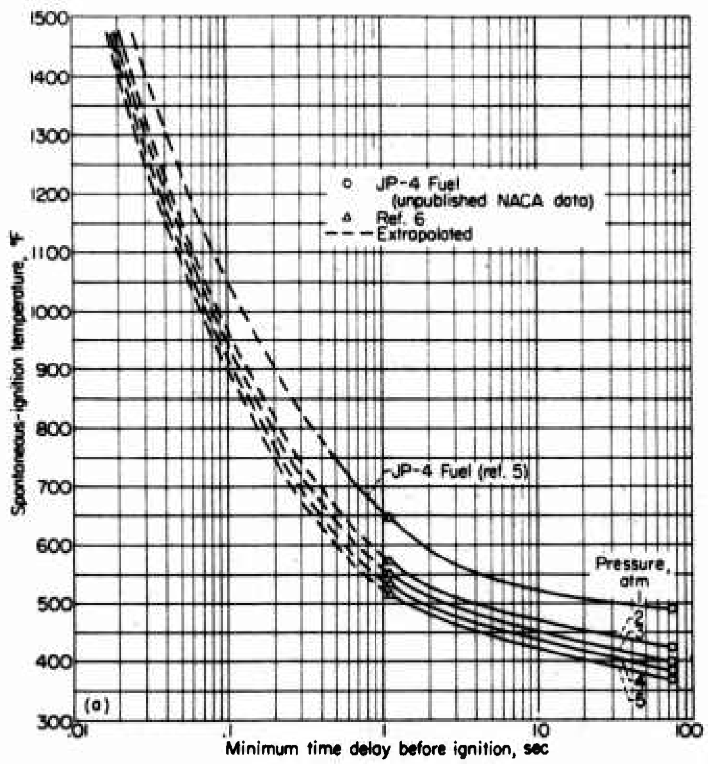

Figure 24. JP-4 Ignition Delay vs. Surface Temperature (From Reference 8).

Two main trends are evident:

a. The required contact time decreases with increasing surface temperature.

b. When the combustible atmosphere is at a higher pressure, which exists within an engine during the initial stages of a crash, ignition is possible at a lower surface temperature for a given contact time.

3. Most of the experimental data on ignition of combustible atmospheres flowing over hot surfaces were obtained with single-constituent hydrocarbon fuels under conditions that hardly approximate those in a reciprocating or jet engine.

4. Assuming all things equal (exposure time, pressure, combustible mixture, etc.), JP-4 has a lower flat-plate ignition temperature than aviation gasoline, and the lower grades of gasoline have lower flat-plate ignition temperatures than the higher grades (reference 14).

5. Reduced fuel volatility would not increase the ignition temperature of the fuel but would reduce the vapor volume in which an ignition source is hazardous.

According to early tests, one authority in the field (reference 14) contends that for complete immunity to fires ignited by the exhaust system, the surface temperatures should not be greater than 500 degrees F. (turbine fuels not considered).

Another authority (reference 9) was of the opinion that cooling the exhaust system to 760 degrees F. would be adequate. This author contends that since oil ignites at a lower temperature than gasoline, it determines the design cooling temperature. Further, he contends that oils and hydraulic fluids would normally, upon delayed spillage, flow by gravity to the lower portions of the engine. Thus, not only would it take time to reach the hot spots, but also it would be in a liquid form rather than in a mist. According to reference 9, oils kept in a liquid state possess a higher ignition temperature than oils in vapor or mist form. Thus, the chances of their being ignited below 750 degrees F. in a liquid form are extremely low. Any oil that hits the manifold during the time that a water-spray cooling system is in operation, and the temperature of the manifold is above 750 degrees F., would be evaporated along with the water and carried away.

After the water spray had terminated, the water would settle in the lower portions of the engine area where the flammable fluids would flow by wetting conduction.

In later tests where turbine fuels were involved (reference 8) an experimental approach to the safe-temperature problem was undertaken. The hot surfaces were simply cooled by a water-spray method, and the duration of the spray was recorded. After each spraying, flammable fluids were sprayed on the hot surfaces. It was soon determined what duration of coolant spray was required, for a given engine location, to reduce the temperature low enough so that when the reheat phenomenon (see page 22) occurred, ignition would not take place.

Based upon the literature reviewed, it was decided to cool the hot surfaces to at least 400 degrees F. in the H-21 fire-inerting system. This would provide a safety factor in the event of poor coolant distribution due to damage to the system during impact or to other causes.

APPENDIX III. PHOTOGRAPHS OF CRASH FLIGHT

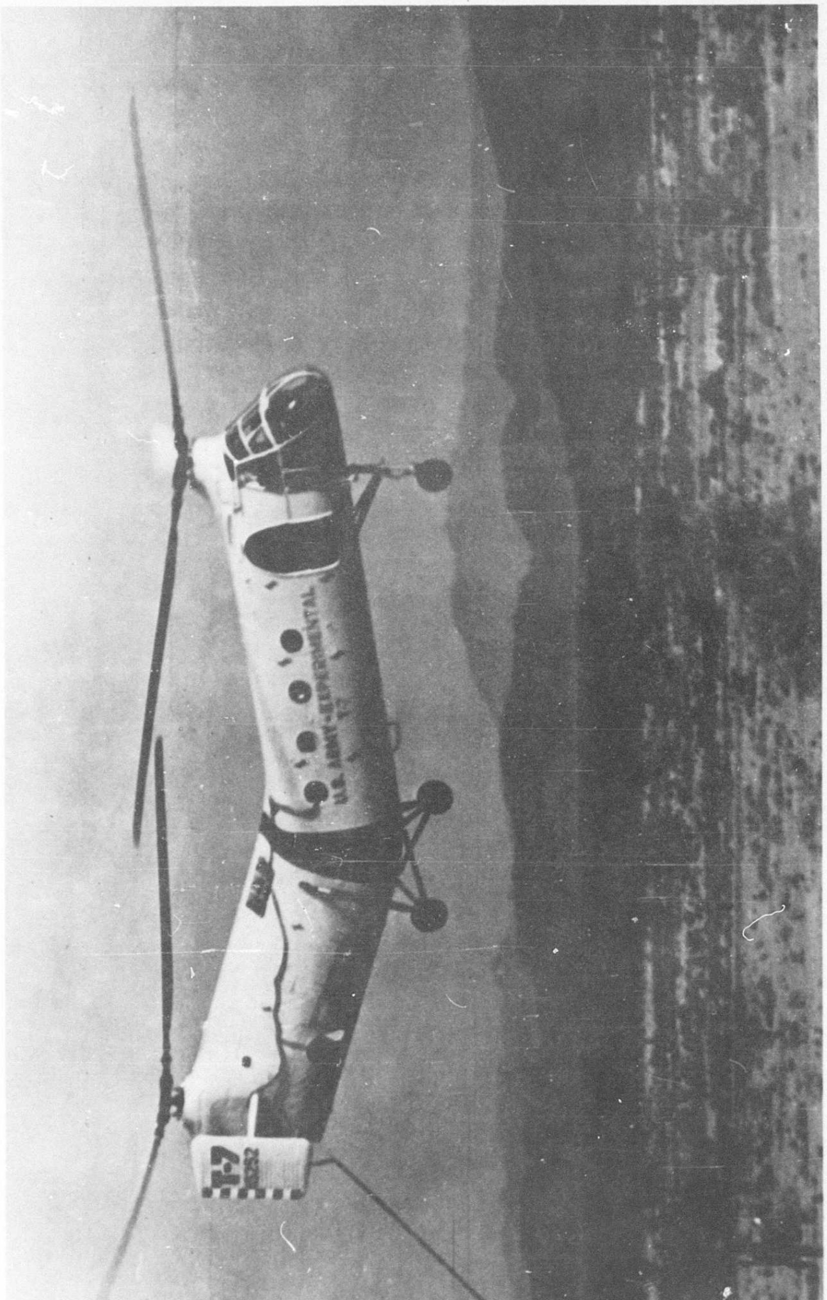

Figure 25A. Photograph shows test vehicle during flight under remote control just before beginning descent phase.

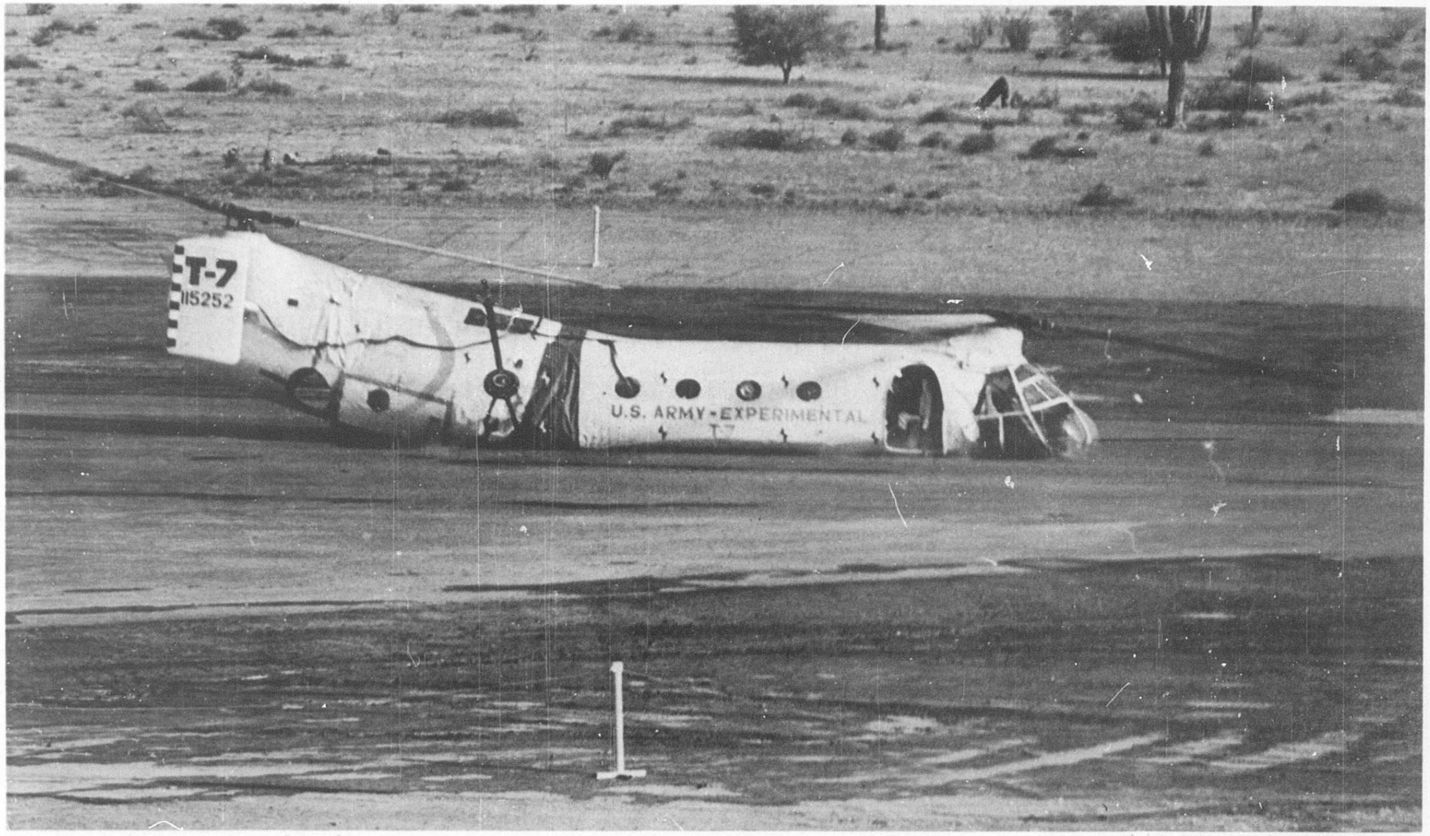

Figure 25B. Photograph shows aircraft just after impact. The nose gear and both main gear have completely failed. Note the wrinkles in the fuselage skin just below the forward rotor between cockpit and right door and also in the dark band just ahead of the right main gear and in the tail section just above the access opening to the engine compartment.

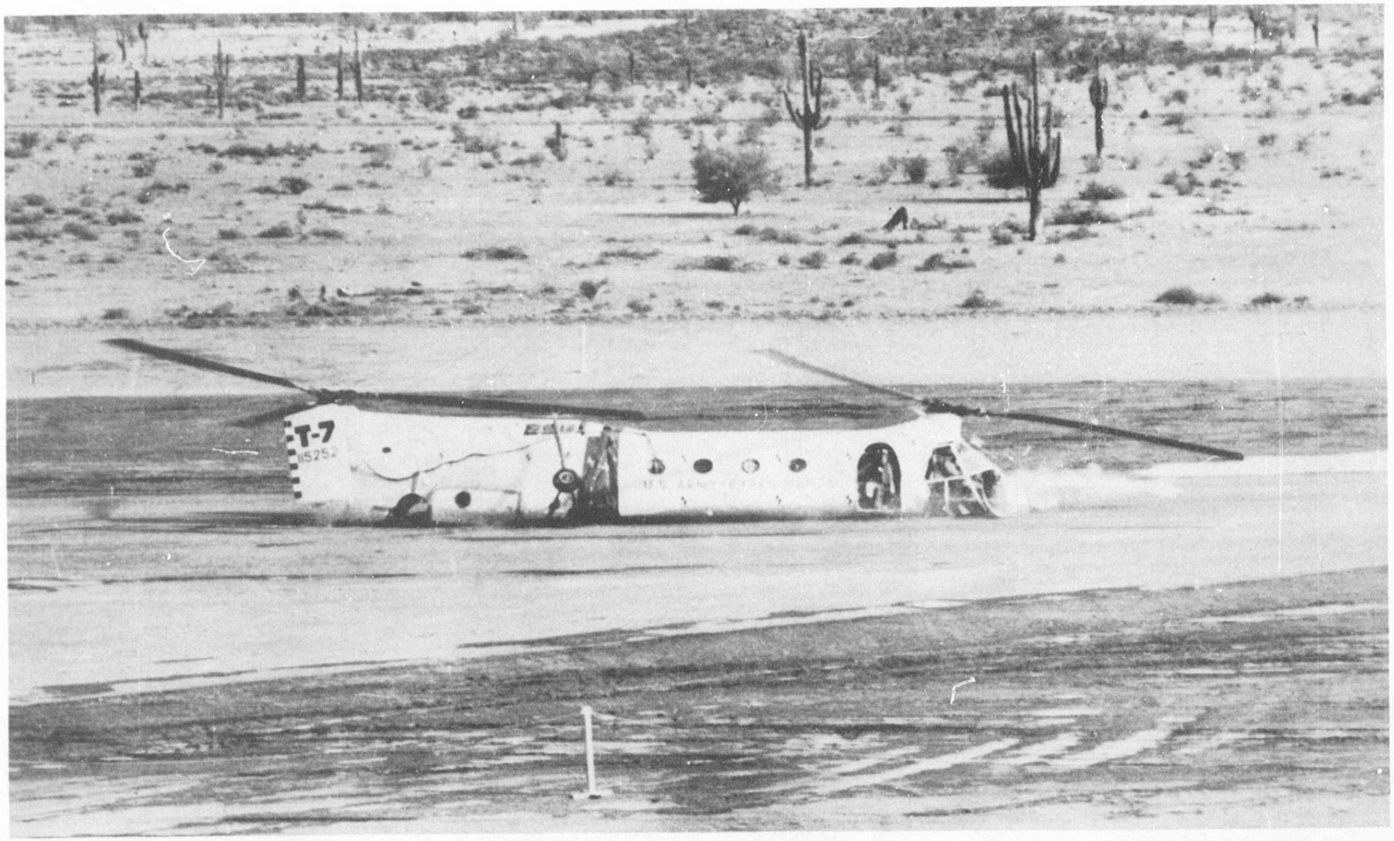

Figure 25C. In this photograph, the aft end of the fuselage has moved downward until the entire bottom surface of the fuselage is in contact with the ground. The fuselage skin is beginning to tear in the dark band just ahead of the main landing gear near the top of the fuselage.

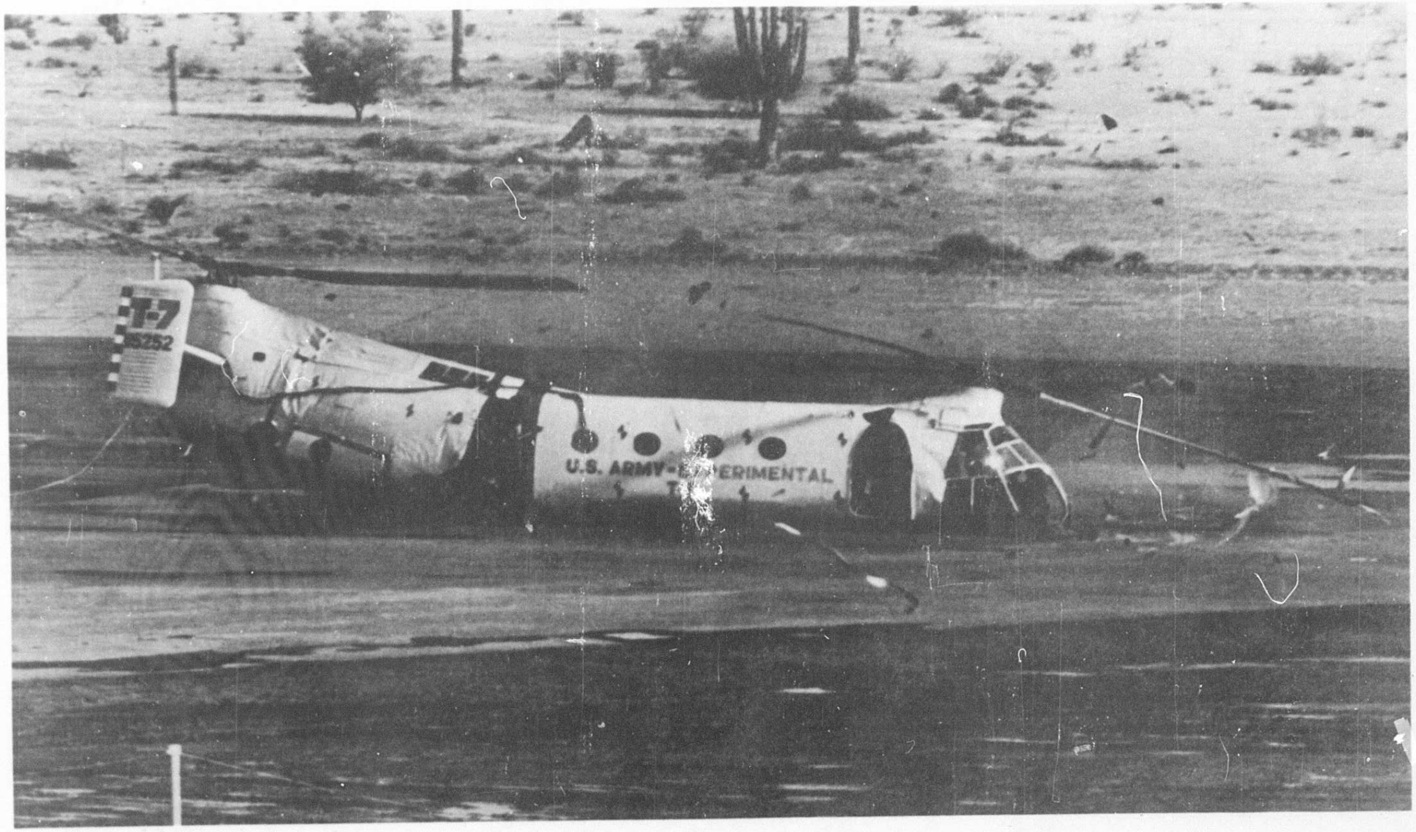

Figure 25D. This photograph shows the rear of the aircraft "bouncing" and the tail moving upward again to its normal position. The dark section of the fuselage has opened considerably. Also, the forward rotor blades have now contacted the ground and are breaking up. Fuel spillage is occurring. Time: + 0.943 Second.

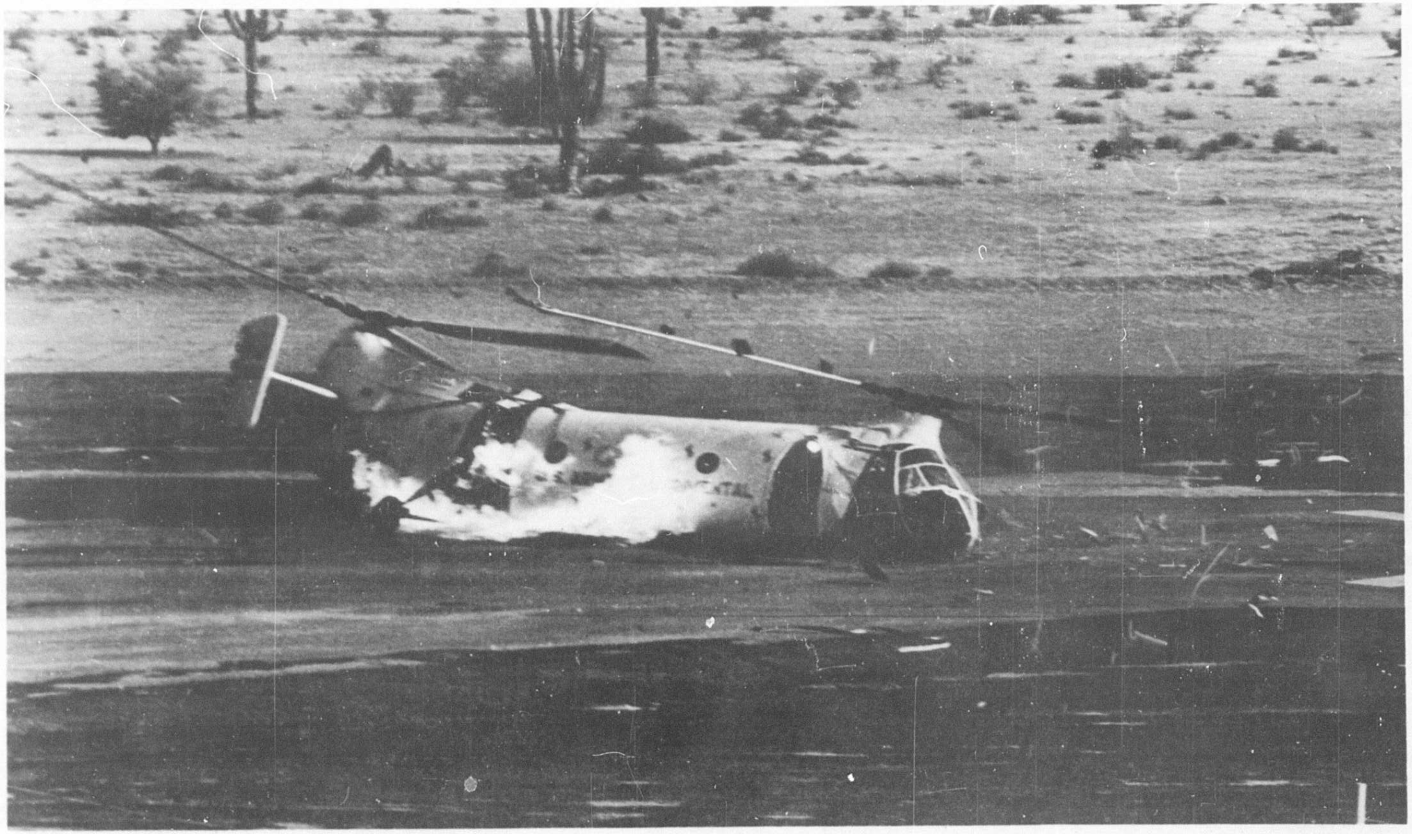

Figure 25E. This photograph was taken shortly after the flash fire began with the aircraft now beginning to rotate clockwise. High-speed films show that ignition occurred approximately at the center of the dark band at ground level.

Time: + 2.032 Seconds.

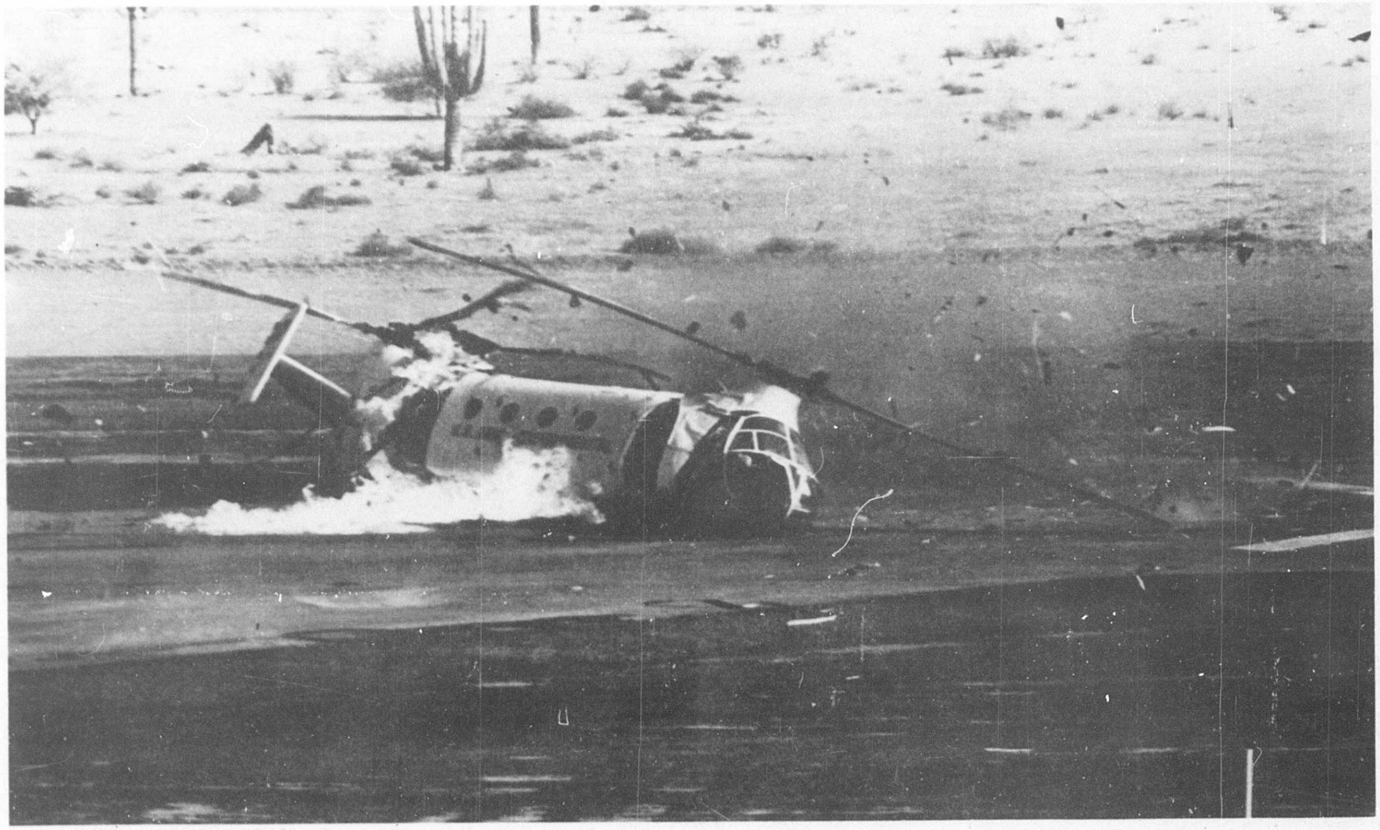

Figure 25F. This picture was taken shortly after both rotors had contacted the runway surface and were breaking up. Note tail section and its relationship to the forward portion of the fuselage.

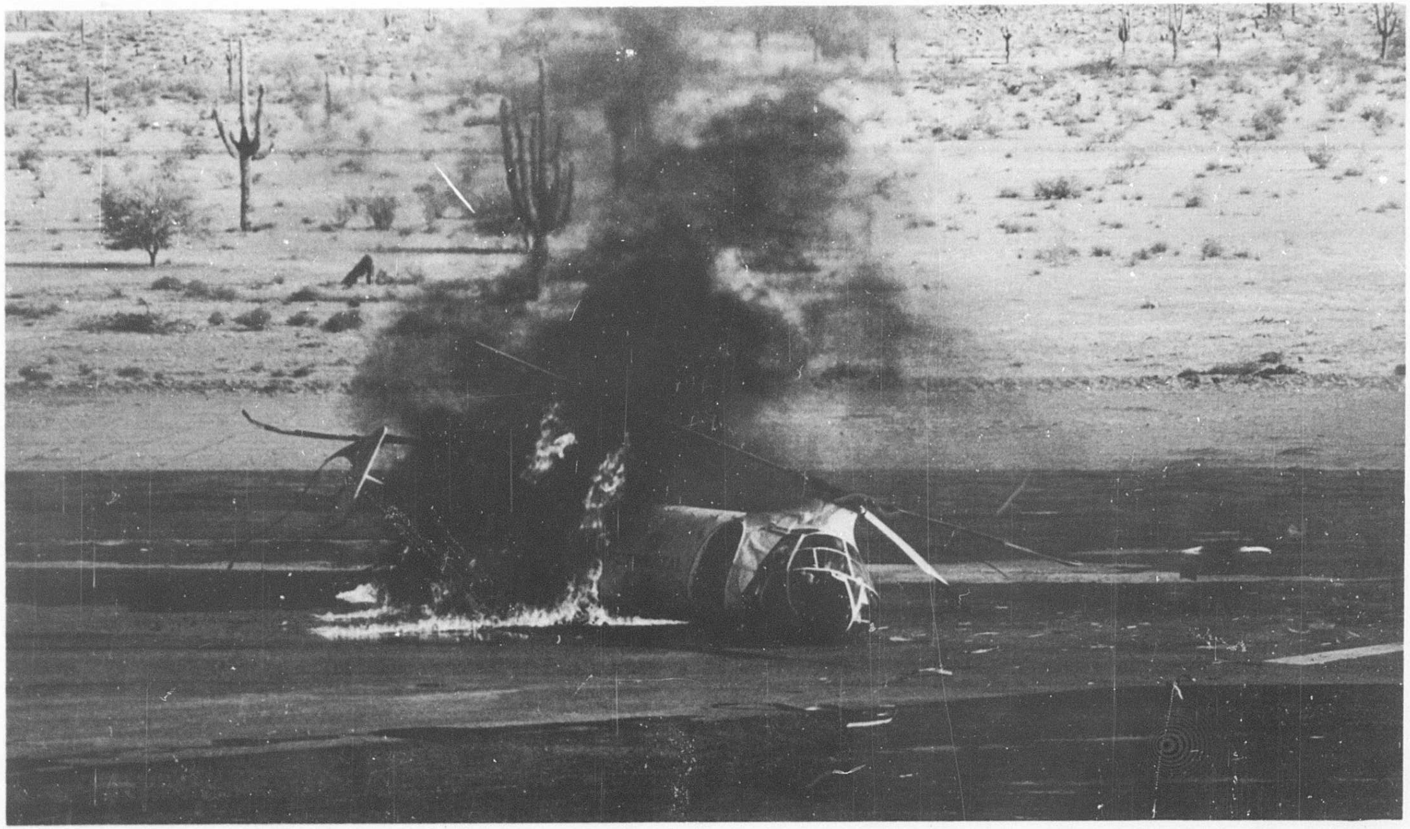

Figure 25G. In this photograph the aircraft has come to rest, and the flash fire is beginning to die down. Time: + 60.0 Seconds

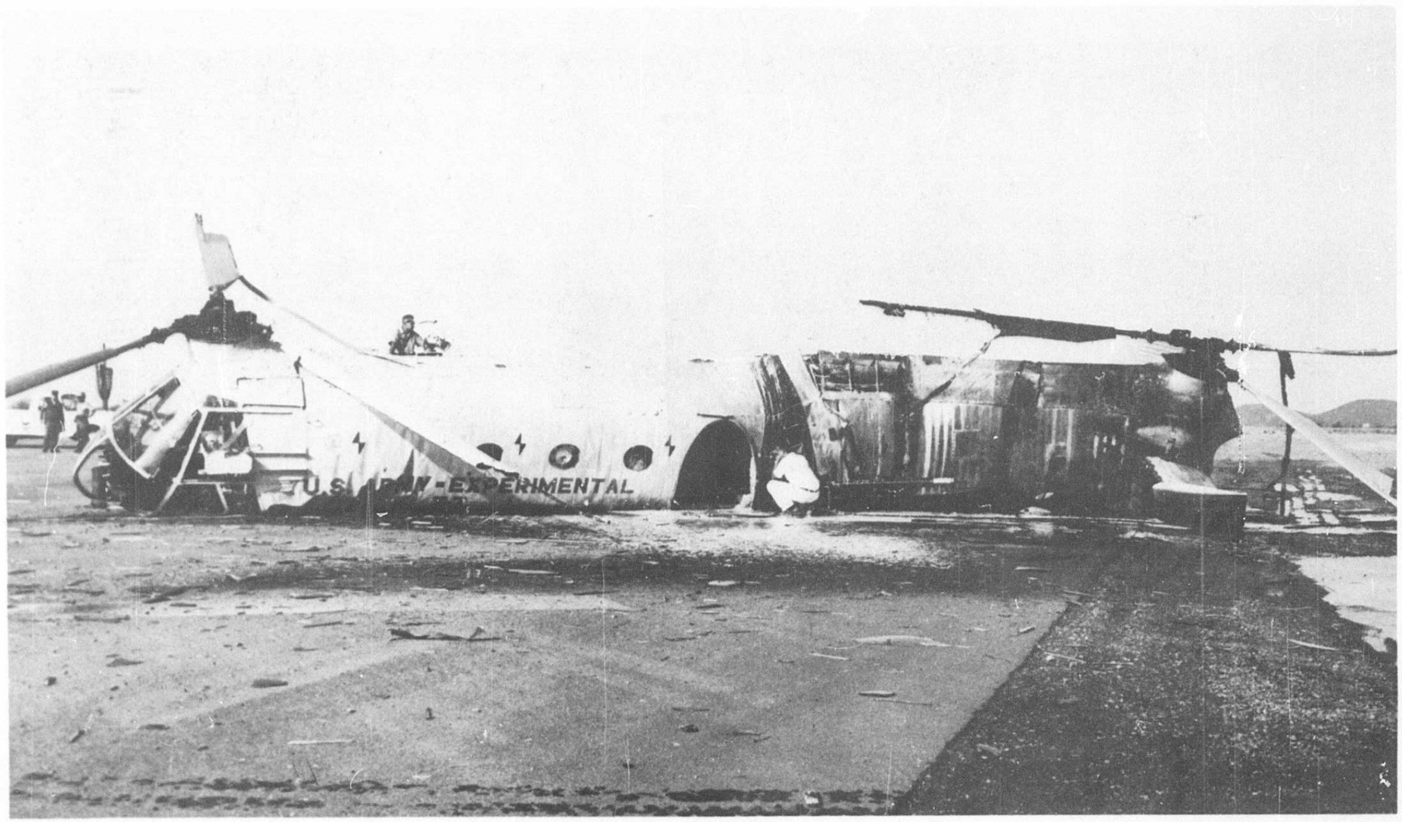

Figure 25H. Postcrash View.

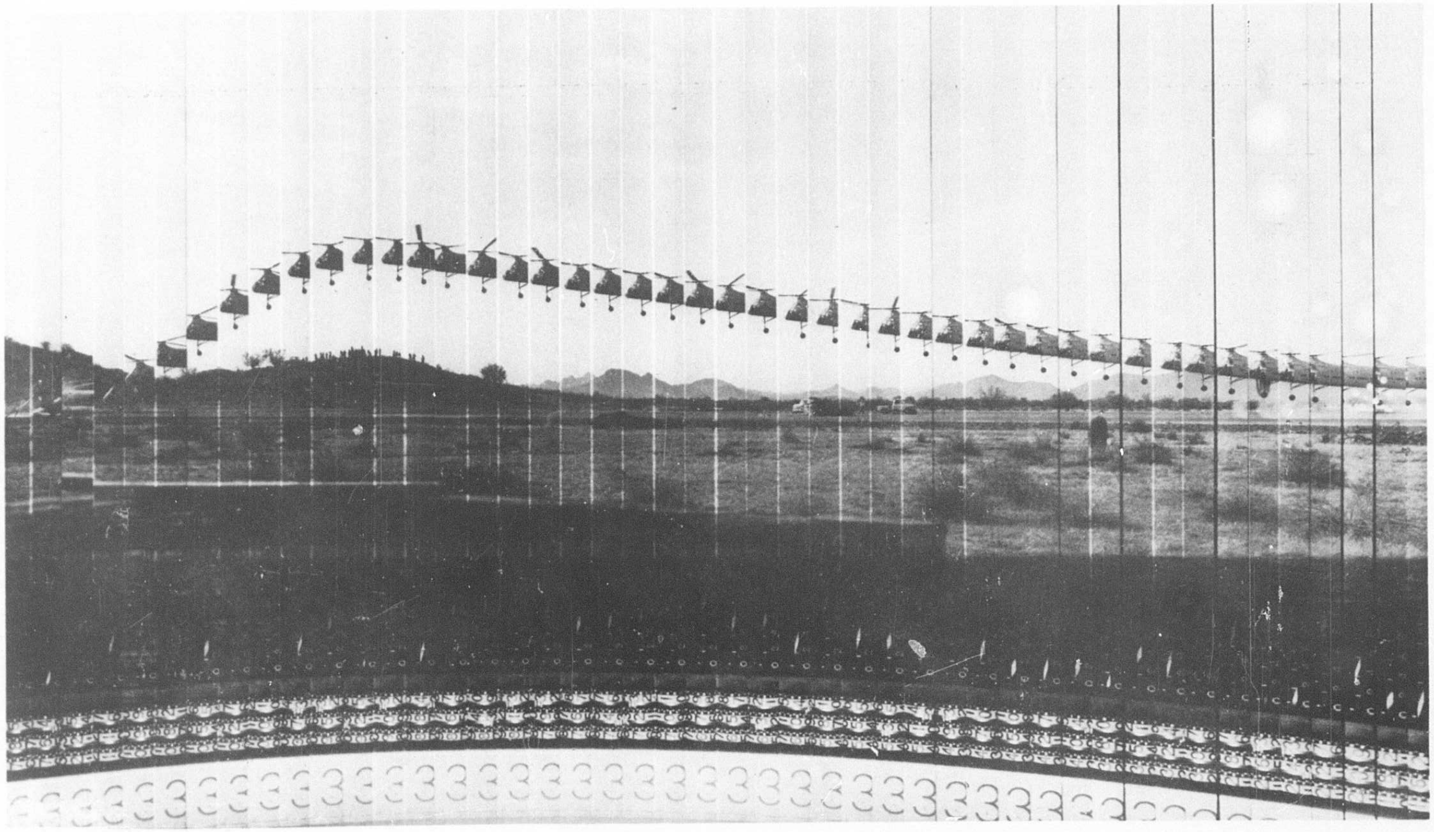

Figure 25 I. This is a print of the Fairchild Flight Analyzer film showing the flight from shortly after takeoff to impact. There is approximately 11 feet of aircraft travel between exposures.

APPENDIX IV. FIRE-INERTING-SYSTEM WEIGHT

	Pounds
Water accumulator	16.00
Nitrogen bottle, regulators, high and low manifolds . . .	16.00
Carbon dioxide bottle, valve, and distribution manifolds .	9.25
Oil cutoff valve	3.25
Fuel and butterfly closing mechanism	1.00
Aspirator, squib valves, and spray manifold	4.00
Connecting flex lines	21.00
Spray distribution tubes	5.00
Nitrogen-water mounting bracket	7.00
Carbon dioxide mounting bracket	1.00
Water	16.00
Carbon Dioxide	4.25
Nitrogen	1.00
	104.75

APPENDIX V.

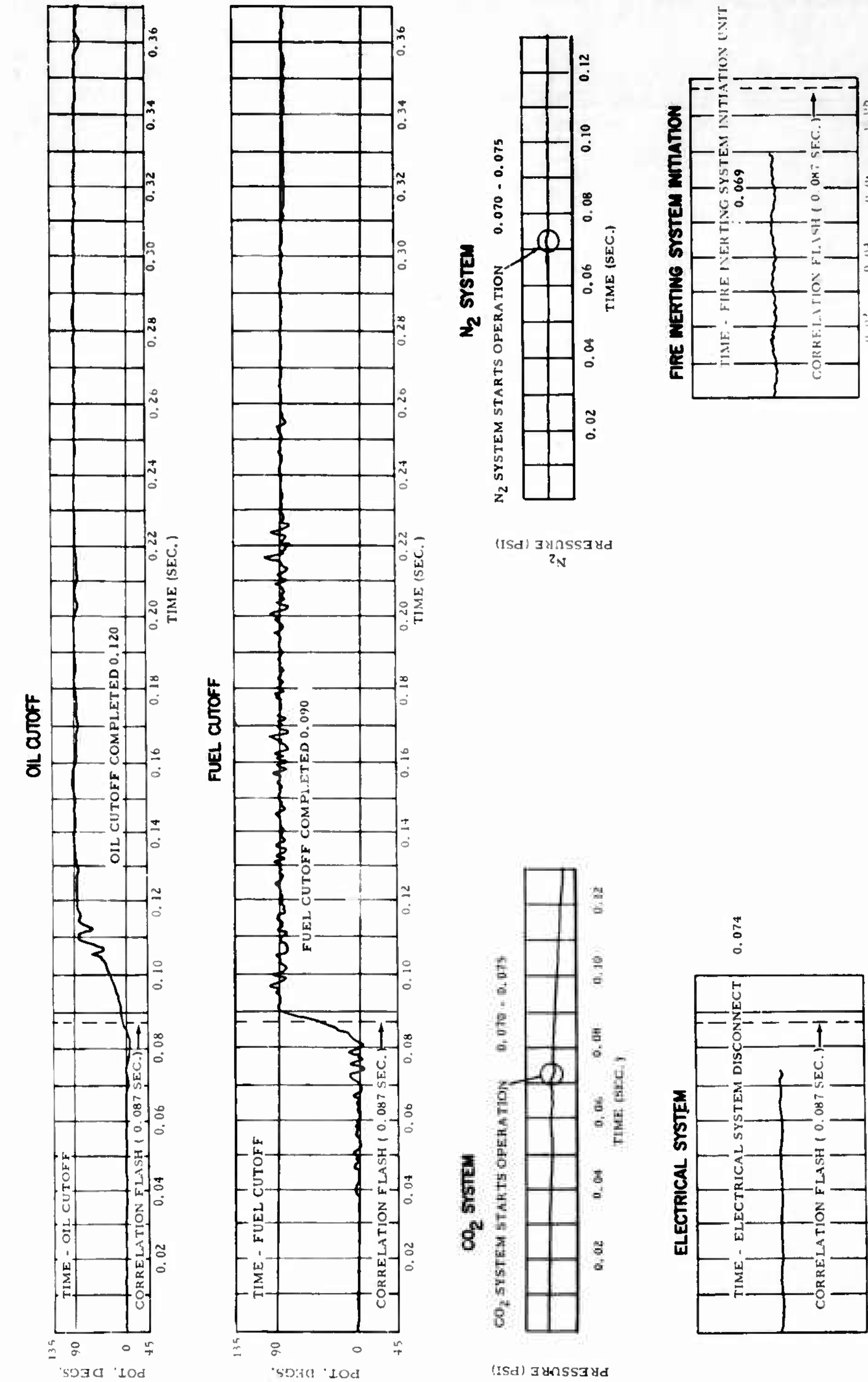

Figure 26. Oscillograph Records (FIS Timing).

COMPENDIUM

Based upon statistical studies (Reference 1 and 2) it appears that, of all Army aircraft which burn following a crash, helicopters have the lowest survival rate. Aviation Safety Engineering and Research (AvSER), a division of the Flight Safety Foundation, was awarded a contract to investigate the problems arising in the development and employment of a fire-inerting system appropriate to the impact conditions occurring in potentially survivable helicopter accidents which normally result in fire.

The literature search which preceded the experimental testing included a thorough evaluation of the work done on postcrash fire prevention in fixed-wing aircraft at the NACA Lewis Laboratories during 1949-1957 and the subsequent work done by the Walter Kidde Company for the Air Force in 1953-1958. The NACA reports clearly demonstrated that fire-inerting systems capable of preventing fires in approximately 80 to 90 percent of the potentially survivable fixed-wing accidents could be based upon the control of only two ignition sources: (1) the aircraft electrical system and (2) the aircraft exhaust system including inerting of both hot gases and hot surfaces.

Postcrash fires occur only when three basic elements are present - a fuel, an oxidizer, and an ignition source. Control of all three elements would be ideal: however, complete control of the oxidizer (the atmosphere in the crash area) does not appear feasible, although local control of the oxidizer at specific ignition points is possible as shown in the work reported by NACA and also by Walter Kidde and AvSER. However, fuel control in the form of fuel containment can effectively reduce fire hazard and fire volume when leaking fuel is ignited from controlled sources such as static or friction sparks. The ignition source appears to be the most feasible of the three elements to control.

In the application of fire-inerting systems to any aircraft, the time-factor is of utmost importance; that is, the system must be activated and must be giving satisfactory inerting at the earliest time at which ignition would normally take place. As a result of the NACA tests, it is believed that for fixed-wing aircraft this is generally about one second or greater. On the basis of impact tests conducted with three H-25 and two H-21 helicopters by AvSER, there is considerable evidence that the time to ignition for accidents involving rates of descent of the order of 30 to 40 feet per second may be as little as 0.20 second. This low time to ignition is associated with the close proximity of the fuel cells to the ignition sources in the helicopter and with the massive fuel spillage

and vaporization occurring in the high vertical G impacts in accidents occurring at these rates of descent. It should be observed that accidents within the 30 to 40 feet per second vertical descent are potentially survivable from the crash injury point of view.

Obviously, a fire-inerting system is practical only if it provides inerting within the time limits imposed by the flight characteristics and configuration of the specific aircraft in which it is installed. In view of the apparent 0.20 second time constant for the H-25 (and H-21) helicopters, AvSER has concentrated its present efforts upon the development and testing of a system applicable to an H-21 aircraft. Emphasis was placed on the inerting of the engine and engine exhaust system, under the hypothesis that the electrical system can readily be inerted in 0.20 second. This fact was demonstrated as a side experiment in the subsequent test of the engine system, although no effort was made to develop a prototype electrical inerting system suitable for installation on a service aircraft. This is considered to be a routine developmental project.

The basic requirements established for the experimental engine inerting system described in this report were:

1. To cool all hot surfaces to temperature below the ignition temperature of flammable fuels and oils.

2. To provide local oxidizer control over all hot surfaces until cooling of such surfaces drops below the ignition temperature of the flammable fluids.

3. To inert the interior of the engine from the induction system to the exhaust system outlets.

4. To shut off fuel and oil supply to the engine.

5. To provide the above functions with a weight limitation of 100 pounds and a time-to-inerting limitation of 0.20 second, when subjected to a crash environment typical of a severe, but potentially survivable accident.

Many experiments, some of which are discussed in the report, were conducted in order to establish the specific requirements of the system to provide the functions described above.

A workable fire-inerting system was designed and installed on an H-21 helicopter. On September 12, 1962, the system was dynamically crash tested when the helicopter was flown into a typical accident by radio control.

This report discusses the design criteria established, the experimental procedures used, and the experimental results obtained; and it gives an evaluation of the various separate systems. Basically all systems functioned as designed, a fact which suggests the feasibility of providing engine inerting in the 0.20 second specified. It is almost certain, however, that fuel did not reach the engine itself within this period, due in part to the small fuel load carried. The aircraft was flown on an auxiliary tank, while the main tank was filled with 200 gallons of water topped with 3-1/2 gallons of aviation gasoline to minimize damage to other experiments in the main cabin in event of fire. Additional experiments will be required to establish that complete inerting can be accomplished in such short time intervals when large quantities of fuel are aboard.

It is of considerable interest to note that, even though only 3-1/2 gallons of spilled fuel were available in the crash, ignition did occur. High-speed photographs show the first flame to be in the vicinity of steel bolts and fittings on the landing gear on the right hand side of the aircraft, where ignition from abraded sparks would be expected. This problem can be eliminated through the use of suitable non-sparking materials on those portions of the aircraft most likely to come in contact with the ground in an impact. The presence of the fire in this test clearly emphasizes the effort which must be made to control fuel, oxidizer, and ignition sources if fire inerting is to be accomplished.

This test may be considered satisfactory from the standpoint of the demonstration of the operational capability of the engine inerting system under impact conditions; however, additional work is needed prior to installation of an inerting system in service aircraft, including the development of:

1. Reliable, lightweight hardware for the engine inerting system.
2. The electrical inerting system.
3. A fail-safe initiating system.
4. Fuel containment.
5. Extension to jet engines.

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This report describes the theory of operation and construction (continued)

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